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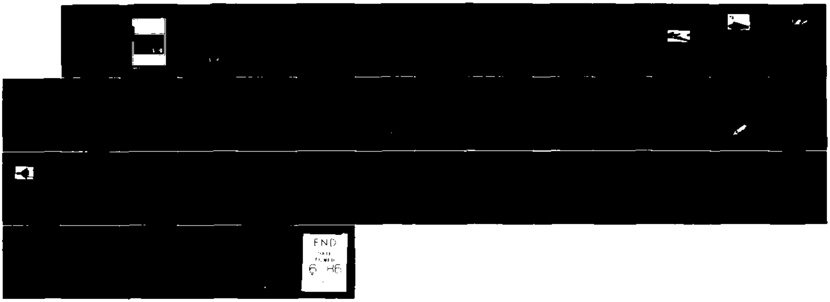

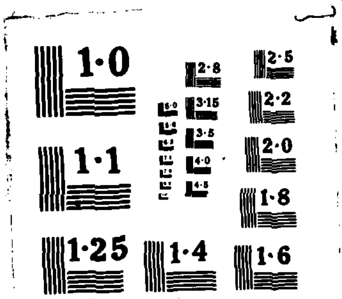
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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

AGARD REPORT No.725

Static Aeroelasticity in Combat Aircraft

NORTH ATLANTIC TREATY ORGANIZATION

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AGARD Report No.725

STATIC *AEROELASTICITY IN COMBAT AIRCRAFT*

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PREFACE

There are several static aeroelastic effects and related problems which are of considerable importance in the design of modern high-performance aircraft. For optimum structural and flight control system design all these static aeroelastic effects must be taken into account in a realistic way, and this demands coordinated effort in several areas of technology.

This publication contains three papers which were presented to the Aeroelasticity Sub-Committee of the Structures and Materials Panel at its 60th Meeting in San Antonio, Texas, USA, 21st—26th April, 1985. All three show that increasing emphasis is now being given to the consideration of static aeroelastic effects in fighter design. There is a clear trend towards the inclusion of static aeroelasticity as a primary design parameter influencing structural optimization, vehicle aerodynamic stability, control effectiveness and overall performance.

H.FÖRSCHING
Chairman, Sub-Committee
on Aeroelasticity

CONTENTS

STATIC AEROELASTICITY IN THE DESIGN OF MODERN FIGHTERS

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Abstract

A review of fighter aircraft development programs over the past ~~three decades~~ (30 years) indicates a trend of increasing emphasis on the consideration of static aeroelastic effects. While early concerns addressed only the impact on air vehicle structural integrity, current design philosophy recognizes and addresses aeroelasticity as a primary design parameter affecting structural optimization, vehicle aerodynamic stability, control effectiveness, and overall performance. Examples from wind tunnel testing, analytical studies, and operational aircraft applications are presented to justify this emphasis, illustrate current methodology and analysis techniques, and make a case for an integrated approach to the consideration of static aeroelastic effects at all stages of the design process.

Introduction

Y. C. Fung, the author of one of the most prominent texts on the subject, defines aeroelasticity in terms of the "phenomena that reveal the effect of aerodynamic forces on elastic bodies" (Reference 1). Another text, Reference 2, refers to aeroelastic phenomena as "the effects, upon the aerodynamic forces, of changes in the shape of the airframe caused by these same aerodynamic forces." Both of these texts make a distinct differentiation between "dynamic phenomena" and the "static aeroelastic phenomena" which the following discussion will be limited to. More specifically, the topic here is the role of static aeroelasticity in the design of modern fighter aircraft.

Static aeroelastic effects are manifested in the form of changes in the total load or lift on the aircraft, or in changes in the overall distribution of load or lift. These changes affect the structural integrity of the vehicle, its static stability, the effectiveness of various control surfaces, and the overall flight performance. The characteristics and magnitude of these aeroelastic effects are dependent on the aerodynamic shape of the vehicle, the structural orientation, the structural stiffness, and the particular flight condition in terms primarily of Mach number and dynamic pressure.

Historically, the study of aeroelasticity began in the early 1920's. However, serious consideration of aeroelastic effects in the design of fighter aircraft, was probably not given until the late 1940's, when significant advances in aircraft performance provided capability for operation at high subsonic speeds and associated dynamic pressures. As indicated in Figure 1, the emphasis on consideration of aeroelasticity has increased over the past three decades. The early 1950's efforts were characterized by minimal consideration, limited to assessing the possible impact on the vehicle structural integrity as a result of overall changes in the vehicle aerodynamic characteristics. Aeroelastic effects were addressed in structural optimization efforts in the 1960's and serious consideration was being given to the impact on performance, relative to control effectiveness and aerodynamic stability. The '70's saw increased emphasis on structural optimization to enhance performance with the advent of serious aeroelastic tailoring and designed-in structural flexibility. Design philosophy today recognizes aeroelasticity as a primary design parameter with dedicated testing and analyses being considered a necessary and integral segment of the vehicle design process.

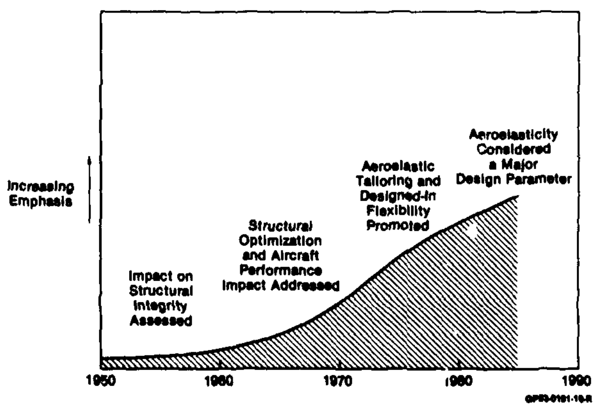

Figure 1. Evolution of Aeroelastic Considerations in Fighter Aircraft Design

Configuration Effects

Lifting surface structural flexibility effects are typically the primary aeroelasticity consideration in fighter aircraft design. Fuselage flexibility is, in general, a secondary consideration. The relatively high density of this structural component, designed to sustain high acceleration levels, and the high structural loadings produced by the close coupled design of most modern configurations, result in high fuselage stiffness. The thin airfoil sections utilized on the lifting surfaces of high speed aircraft, however, lead to inherently flexible structural components with potential aeroelastic sensitivity. Two typical fighter aircraft wing planforms are presented here to illustrate the effects of planform geometry and associated structural orientation on resultant aeroelastic characteristics. The relatively unswept configuration in Figure 2 is basically torsion sensitive. The deflected shape under a subsonic aerodynamic loading exhibits a divergence characteristic as shown in Figure 3. Only overall panel stiffness is considered, and an effective elastic axis, for a conventional structural concept is assumed at 40% of the local chord. The pressure loading and the non-dimensional lift distributions for both the rigid and flexible cases are presented in Figures 4 and 5. The lift distributions illustrate the structural-load-magnification aeroelastic characteristic. Structural optimization of this wing must provide adequate stiffness to insure a divergence speed well in excess of the operational envelope of the aircraft. The swept wing in Figure 6 exhibits bending aeroelastic sensitivity due to the orientation of the main structural torque box. Note the highly swept outer panel reference axis. In Figure 7, the deflected shape of this wing under a maneuvering load, illustrates the swept-axis-bending induced streamwise twist. The transonic loading illustrated in Figure 8 tends to accentuate the aeroelastic relief due to the relatively far aft chordwise center of pressure on the cambered airfoil. The lift loss in the area of the wing tip is apparent in the comparison of rigid and flexible spanwise distributions in Figure 9.

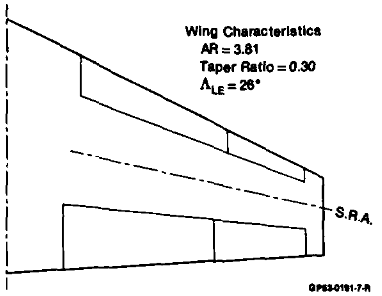

Figure 2. Unswept Wing Configuration

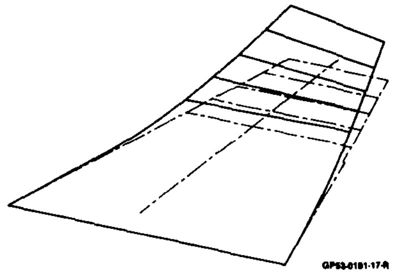

Figure 3. Unswept-Wing Deflection Characteristics

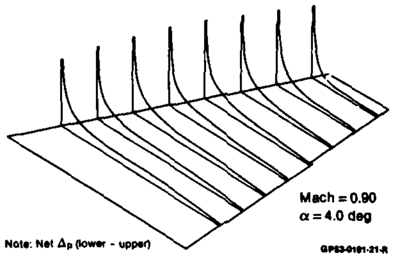

Figure 4. Typical Subsonic Pressure Distribution on an Unswept, Uncambered Wing

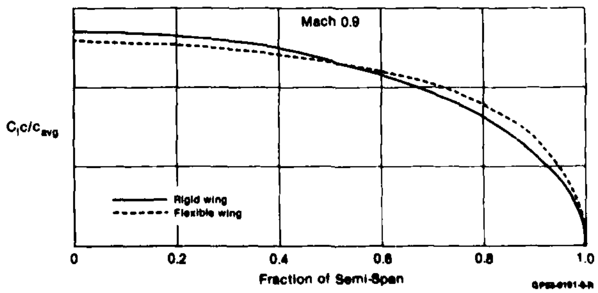

Figure 5. Unswept Wing Nondimensional Spanwise Lift Distribution

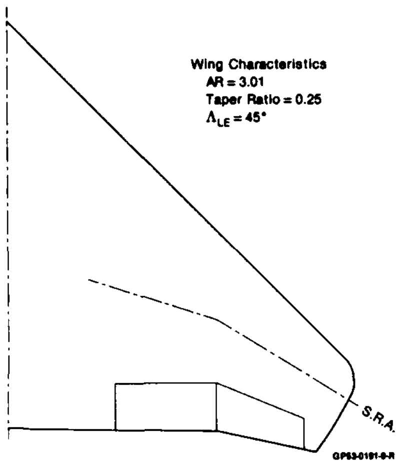

Figure 6. Swept Wing Configuration

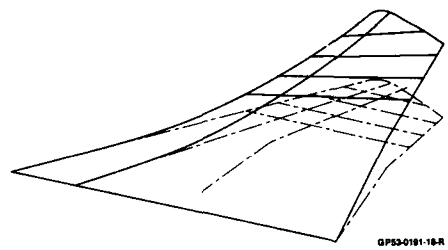

Figure 7. Swept-Wing Deflection Characteristics

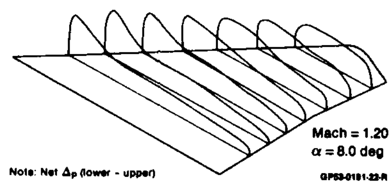

Figure 8. Transonic Pressure Distribution on a Swept Cambered Wing

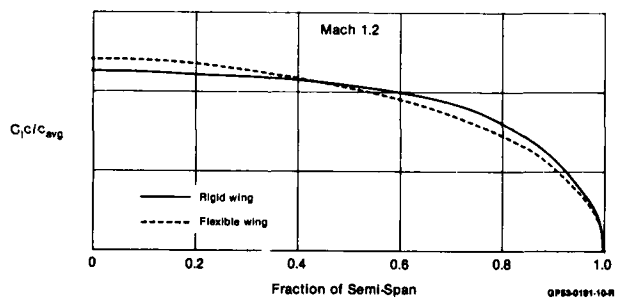

Figure 9. Swept Wing Nondimensional Spanwise Lift Distribution

The data presented above has all been derived by analysis using generally accepted lifting surface aerodynamic codes. Analytical aerodynamics currently provides the foundation for the majority of our aeroelastic design activities and configuration optimization efforts. Two areas of investigation are not being adequately addressed with analytical tools, however. Current state-of-the-art aerodynamic codes do not provide sufficient accuracy to predict either local flow anomalies in the transonic flight regime or the non-linear effects of flow separation observed at elevated angles of attack. The Euler and Navier-Stokes formulations (References 3 and 4) and iterative perturbation techniques (Reference 5) are producing promising results. Advances in computer technology may allow routine use of these complex codes in the future design environment. Wind tunnel testing is currently required, however, to obtain accurate data in these areas.

Wind Tunnel Testing

Figure 10 illustrates a model used in a limited aeroelastic wind tunnel investigation performed in the mid-1960's. The variable sweep configuration is defined in more detail in Figure 11, and the construction technique employed for the flexible wing panels is shown in Figure 12. The welded steel skeleton was packed with polyurethane foam and encased in silicone rubber which provided the appropriate surface contour. The strain gage instrumentation located near the wing root was calibrated to measure panel shear, bending moment, and torsion. Testing was performed at subsonic and supersonic Mach numbers and various dynamic pressures to determine aeroelastic effects on loads and on aircraft stability. Test conditions were duplicated with a set of rigid wing panels to establish a base. An additional objective of the flexible model testing was to obtain correlation data to validate or refine lifting surface aerodynamic codes. Figure 13 illustrates the correlation between predicted variations in total model normal force and wing panel root bending moment with dynamic pressure. Good agreement is shown at this subsonic Mach number and low angles of attack. Figure 14 illustrates the normal force and pitching moment characteristics obtained from the model main balance. Note the unstable break in the rigid model pitching moment and the lack of a dominant break in the flexible model data. The apparent delayed wing tip flow separation on the flexible model was also reflected in the wing bending data. This singular example provides a strong case for aeroelastic wind tunnel model testing. Linear theories would not have predicted the rigid model stability and, with appropriate rigid wind tunnel model data available, linear theory would not provide the appropriate aeroelastic corrections.

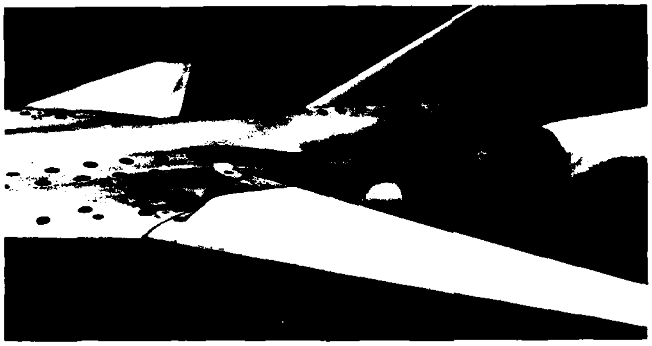

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Figure 10. Variable Sweep Flexible Wing Wind Tunnel Model

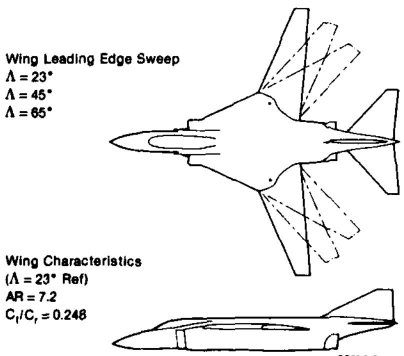

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Figure 11. Wind Tunnel Model Configuration

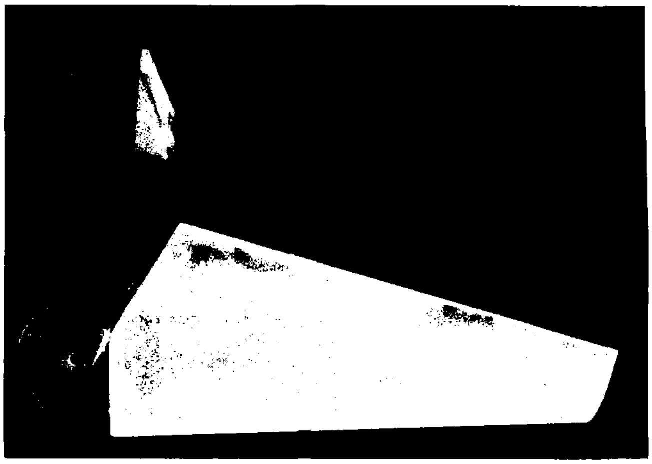

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Figure 12. Wind Tunnel Model Flexible Wing Panel Construction

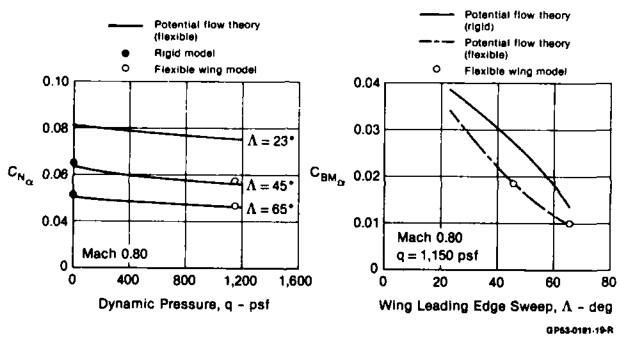

Figure 13. Effect of Wing Sweep on Lift and Lift Distribution

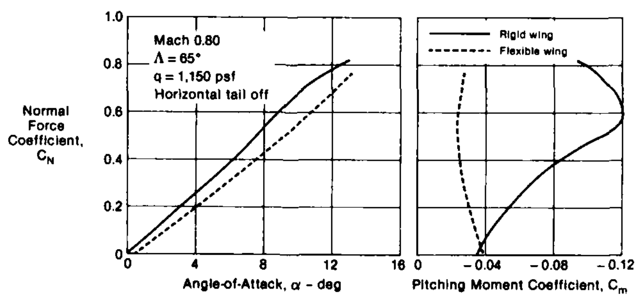

Figure 14. Lift and Pitching Moment Characteristics

Aeroelastic panel construction techniques employed in other test programs are illustrated in Figure 15. Approach (b), with a stiffness scaled beam machined along a predicted elastic axis, and load isolation cuts forward and aft of the beam, has proved to be most successful. Approach (a), with a foam filled steel skeleton and fiberglass covering was an attempt to reduce the mass and improve the model flutter margin. The minor improvement achieved in testing range did not justify the added complexity of the model. Approach (c) employs a multi-layer fiberglass layup and may be appropriate for small surfaces. However, stiffness distribution control is difficult with this type of construction.

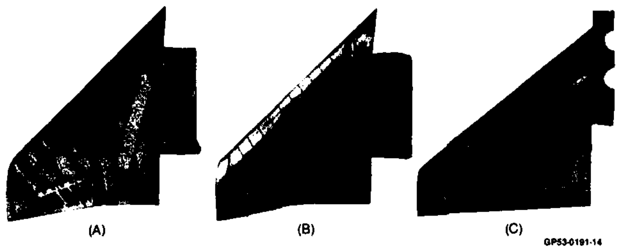

Figure 15. Aeroelastic Wind Tunnel Model Construction Techniques

Analysis Techniques

As mentioned above, the bulk of the aeroelastic design and evaluation effort is supported by analytical methods. Recent improvements in the various aerodynamic theories and panel aerodynamics computer codes have provided a source for an appropriate loading definition throughout a large portion of the operating envelopes of current and projected future fighter aircraft. The other major component of the aeroelastic analysis is the structural stiffness model. Current structural design methodology employs the finite element modeling techniques of computer programs such as NASTRAN. The optimization capabilities and inherent comprehensive accuracy of these techniques have resulted in a dependence on their utilization in virtually all phases of the design process. A by-product of the finite element internal loads solution is an accurate and highly detailed structural stiffness definition which may be used directly in the aeroelastic analysis. Figure 16 is a point-line representation of the major elements in a typical wing finite element model. The bold points indicate locations at which influence coefficients would be obtained to provide a comprehensive stiffness representation of the panel. An aeroelastic analysis utilizing a representation of this type and an appropriate aerodynamic theory will yield not only a detailed definition of the net loading, but also a complete deflection pattern for the loaded structure as illustrated in Figure 17.

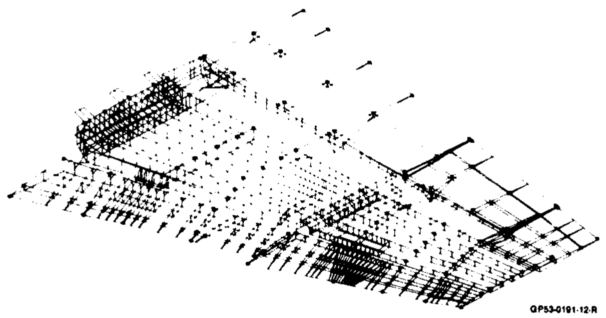

Figure 16. Wing Finite Element Structural Model

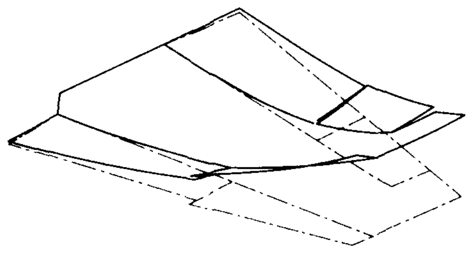

Figure 17. Deflected Shape of Wing Structural Model Under a Typical Maneuvering Loading

An evaluation of the aeroelastic implications of minor structural concept changes, or parametric tradeoffs on a baseline configuration, may require utilization of a simplified, more versatile, stiffness representation of the structure. An effective beam representation of the total panel stiffness is generally applicable and appropriate for these needs and also satisfies the requirement for defining wind tunnel model construction. Model strength and scale factors typically dictate a beam stiffness approach to the representation of the full scale surface flexibility.

Figure 18 presents the results of a control effectiveness study performed on a desk top computer utilizing an effective beam stiffness model. The swept wing configuration of Figure 6 exhibits torsional aeroelastic sensitivity, as well as primary bending sensitivity, when loaded by a deflected trailing edge aileron. At high dynamic pressure flight conditions, the flexible wing loading produces the zero net aileron effectiveness, or actual aileron reversal, as illustrated in Figure 18. Structural stiffness parametric studies are easily performed on the simplified beam model by factoring the effective bending stiffness (EI) or torsional stiffness (GJ) to produce the results shown in Figure 19.

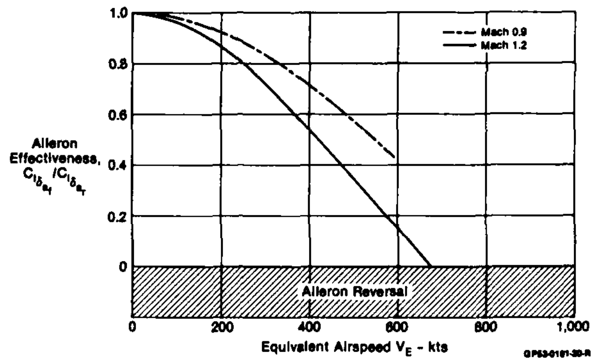

Figure 18. Typical Aileron Effectiveness Trends

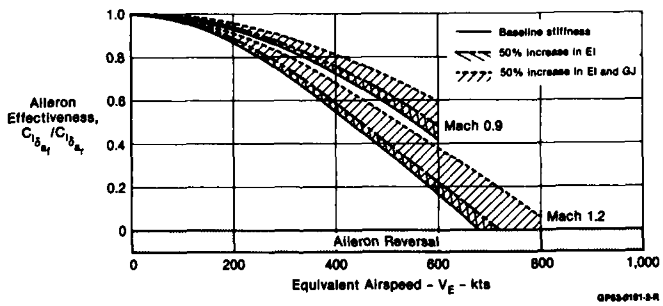

Figure 19. Parametric Study of Aileron Effectiveness vs Stiffness

Operational Aircraft Applications

A fully integrated design environment, in which aeroelastic implications are considered at all stages of the design evolution, provides the capacity for implementing aeroelasticity-dependent design features to enhance the overall performance of the evolving configuration. Early identification of an aileron effectiveness deficiency in the configuration shown in Figure 2, for example, could lead to incorporation of an aeroelastic device to enhance the roll power. The leading edge flaps of this wing are designed to improve the low speed, high lift, or high angle of attack characteristics of the thin, sharp edged airfoil. Deflection of the flaps at subsonic Mach numbers on a rigid wing produces the characteristic chordwise pressure profile shown in Figure 20. The net loading effect is to produce zero wing lift, but a large leading-edge-up wing torque. Aeroelastically, a significant wing lift is generated as the wing is twisted by the applied torque. A large aircraft rolling moment is generated by deflecting the flaps differentially, left/right. This active aeroelastic performance enhancement device is currently utilized on the F-18 aircraft.

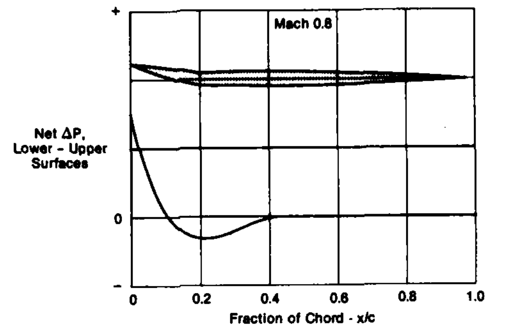

Figure 20. Chordwise Pressure Loading Due to Deflected Leading Edge Flap

The swept wing planform of Figure 6 is implemented on the F-15 aircraft with conical wing camber to enhance the lift characteristics at the primary maneuvering design point. At high transonic Mach numbers and high maneuvering angles of attack, the chordwise load distribution near the wing tip is nearly uniform as illustrated by the rigid model pressure data in Figure 21. High speed roll control on this aircraft is principally achieved from differential deflection of the horizontal tail panels. Due to the aeroelastic bending sensitivity of the wing, the ailerons are relatively ineffective at these conditions. Considering these facts, and the fairly low aileron hinge moments required for subsonic maneuvering, the hinge moment capability of the aileron was judiciously chosen to allow the surface to "float," or unload in high speed, high angle of attack maneuvers. This passive aeroelastic device effectively reduces critical structural loads in the outer panel of the F-15 wing.

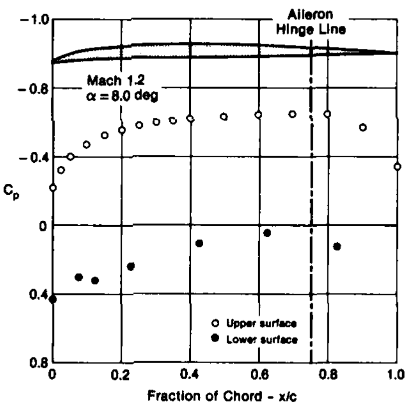

Figure 21. Chordwise Pressure Distribution Due to Symmetric, High Load Maneuvering

Summary

Consideration of static aeroelasticity in the design of fighter aircraft has evolved over the past three decades from a defensive posture to a positive approach of integrated analyses and designed-in structural flexibility to achieve enhanced performance. This positive approach requires coordinated efforts in several technology areas including analytical aerodynamics, wind tunnel testing, structural modeling, aircraft performance appraisal, configuration design, and systems integration. Both active and passive aeroelastic design features have been incorporated in current operational fighter aircraft. Advances in materials, structural concepts, and controls technology are providing expanded opportunities for implementation in the next generation of aircraft.

References

1. Fung, Y. C.: An Introduction to the Theory of Aeroelasticity, Dover Publications, 1969.

2. Etkin, B.: Dynamics of Flight - Stability and Control, Wiley & Sons, 1959.

3. Verhoff, A. and O'Neill, P. S.: A Natural Formulation for Numerical Solution of the Euler Equations, AIAA-84-0163, 1984.

4. Cosner, R. R.: Relaxation Solution for Viscous Transonic Flow about Fighter-Type Forebodies and Afterbodies, AIAA-82-0252, 1982.

5. Gilmer, B. R. and Bristow, D. R.: Analysis of Stalled Airfoils by Simultaneous Perturbations to Viscous and Inviscid Equations, AIAA-81-1239R, 1981.

STATIC AEROELASTIC EFFECTS ON HIGH-PERFORMANCE AIRCRAFT

by

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SUMMARY

Static aeroelastic effects on high performance aircraft may influence the aerodynamic and structural design considerably. Therefore it is important to begin ~~with~~ aeroelastic design studies in the preliminary design phase, ~~to~~ find out problem areas and to develope a general understanding. Structural optimization programs are very helpful in analysing and solving aeroelastic problems on aircraft components. Many papers deal with the activities in the field of structural optimization. Using fiber composites, a new design technique called 'aeroelastic tailoring' provides new capabilities in fullfilling stiffness requirements for aerodynamic surfaces. ~~The~~ benefits of aeroelastic tailoring ~~will be~~ shown on a fighter fin design.
~~The~~ design of advanced digital flight control systems needs a good knowledge of the elastic aircraft behaviour such as elastic derivatives and control surface effectiveness.

It will be shown how structural computer models of a total aircraft can be built up according to the actual development phase, without loosing transparency because of the complexity of large finite element models.

Test-to-analysis comparisons are necessary ~~and possible in many ways~~ in aerodynamics, structures and structural dynamics to up-date computer models and to find out shortcomings of analyses.
Only a combined understanding of different basic principles enables us to apply new technologies to a future project in a benficial and manageable way.

INTRODUCTION

The growing application of new technologies to a high performance aircraft has increased the complexity of the total aircraft system.
To get out most of all individual benefits in an actual aircraft design, it is essential to perform this kind of analysis which combines in a more or less generalised form total aircraft behaviour of different disciplines.

A typical example for such kind of analysis we have in structural dynamics with the very complex field of aeroservoelasticity. This field combines aerodynamics, structures and flight control system in such a way that the interaction of structure to flight control system can be analysed.
Active control technology deals with the design of flight control systems which is based on a given structure and aerodynamics.

In the past time the main task of static aeroelastic was the analysis of correction factors for aerodynamic derivatives due to structural flexibilities.

Modern structural analysis and optimization programs made it possible, to develope structural design proposals to achieve a required aerodynamic behaviour.

During the design and development phase of a high performance aircraft the work in static aeroelastic can be split into the following work packages:

- Definition of aeroelastic design criteria based on preliminary flight control and aeroelastic investigations to detect static aeroelastic effects
- Aeroelastic tailoring and structural optimization of aerodynamic surfaces and work out structural design advices
- Preliminary total aircraft analysis to provide flexibility terms and functions for rigid aerodynamic derivatives

- Test-to-analysis comparisons in structures and aerodynamics according to the actual development phase on models or real structures are essential to match short comings in theorie and computer models.

- Findly the comparison of flight test datas with up dated check analysis

STATIC AEROELASTIC EFFECTS

Performance predictions of new highly manoeuverable fighter aircraft are strongly influenced by the benefits of advanced technologies in aerodynamic configuration, structural design and digital flight control systems. During various design and development phases all these technologies will be combined with their mathematical models in a total aircraft simulation, see Fig. 1

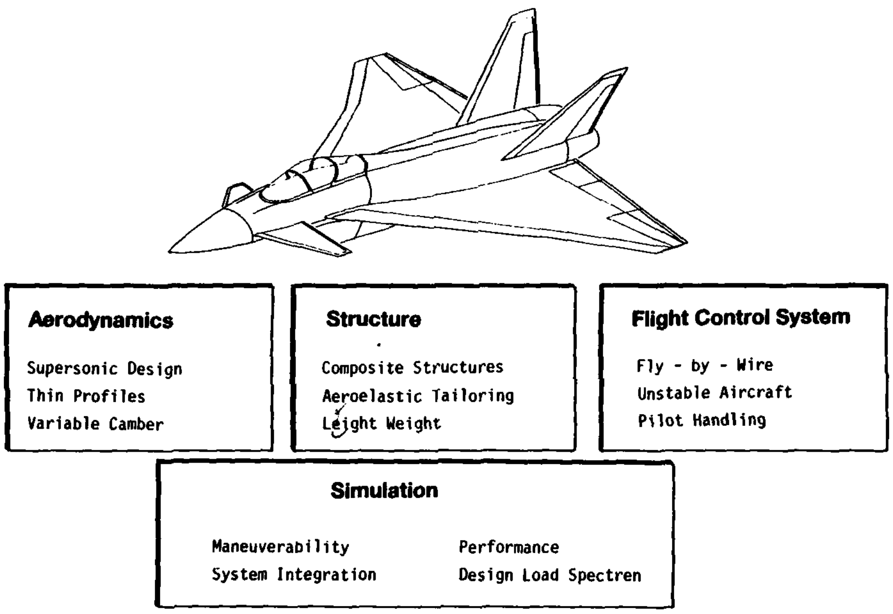


FIG. 1 HIGH PERFORMANCE AIRCRAFT DESIGN

Simulation has a great potential in complex system integration and it is very important to describe the structural behaviour of the airframe in a reasonable way during all design phases. Sometimes it is much better to have a first coars mesh structural modelquickly, than a very fine mesh model to late for the first simulation.

For a future fighter aircraft the preferred aerodynamic design concept is a wing-body-foreplane configuration which promises to meet supersonic performance requirements.

Thin profiles, required leading and trailing edge flap geometry for variable camber concept are important geometry constraints for the wing box. Experience from other projects have shown how structural flexibilities influence aircraft aerodynamics. Therefore an early feetback of elastic behaviour of the airframe to the 'rigid' aerodynamic design is essential to define design criteria and lifting surface stiffness requirements. Aeroelasticity has also a great effect on the flight control system design. If the basic control powers are reduced by increasing dynamic pressure an unstable aircraft will response in undesirable motions, up to an actual loss of control. Therefore the general objective for each aircraft design is to ensure that the aircraft has adequate stability and controllability within the required flight envelope, see Ref. 1.

That means, all aerodynamic datas, coefficients and derivatives for longitudinal and lateral flight mechanic freedoms must be corrected by flexibility terms derived from static aeroelastic calculations. These flexibility termes are necessary for the definition of design criteria for fin, foreplane and front fuselage structures and to up-date preliminary design loads. The problem of the structural design analyst is to provide enough strength for design load cases and to fullfill aeroelastic design criteria.

Many investigations about structural design concepts have revealed a growing usage of fibre/epoxy composites. Aeroelastic tailoring intends the most effective use of anisotropic material to couple both aerodynamic and structural design constraints. A minimum weight structure can only be achieved by an engineered application of structural optimization programs.

AEROELASTIC DESIGN STUDIES

General

Modern computing capacity with aeroelastic analysis and optimization programs and graphic systems enables the engineer to begin his work already in the preliminary design phase. If basic aircraft datas are provided by the project office a first aeroelastic analysis loop can be initiated. In Fig. 2 a geometry study of a wing-fuselage-intake is plotted by a three dimensional graphic system. Such a system is very helpful as a geometry data base for anybody who needs this kind of information.

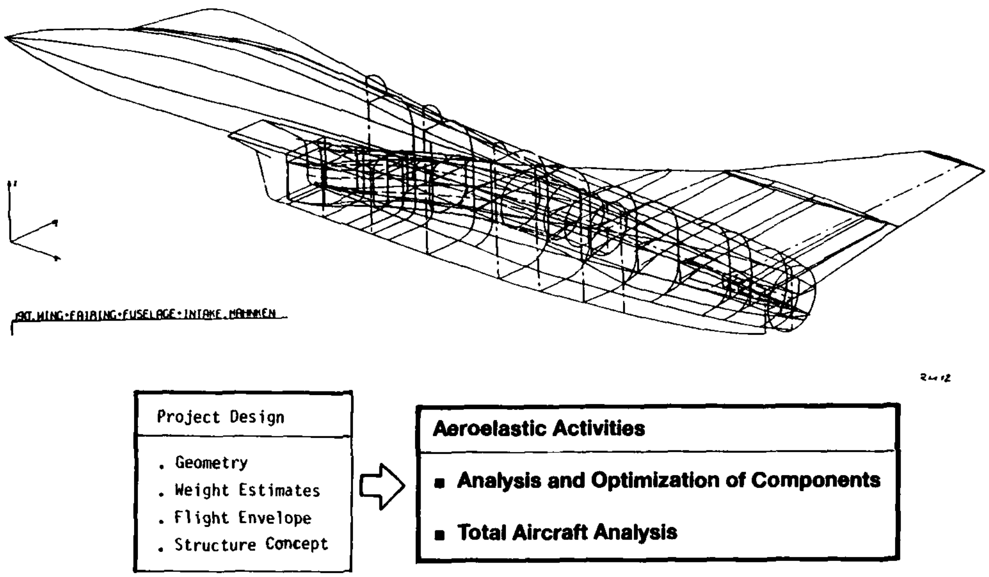


FIG. 2 BASIC DATAS FOR AEROELASTIC DESIGN STUDIES

Aeroelastic activities are now devided into two main parts. Part one will be the analysis and optimization of different aerodynamic surfaces such as, wing, fin and foreplane. The second part will be the total aircraft analysis using all results of various component analyses. A well known procedure which is intended to address the 'total design' problem by combining aerodynamic, static aeroelastic, flutter and stress analysis is the General Dynamics TSO Program (Aeroelastic tailoring and structural optimization procedure). Many papers are written about TSO, see Ref. 2, 3, 4, 5, and therefore it is not necessary to discuss further details.

One major advantage of TSO is the reduction of required input datas to a minimum. The structural computer model for a lifting surface is a plate type structure with trapezoidal geometry and the aerodynamic computer model of the surface might have also a simplified representation of fuselage with foreplane or horizontal tail, see Fig. 3.

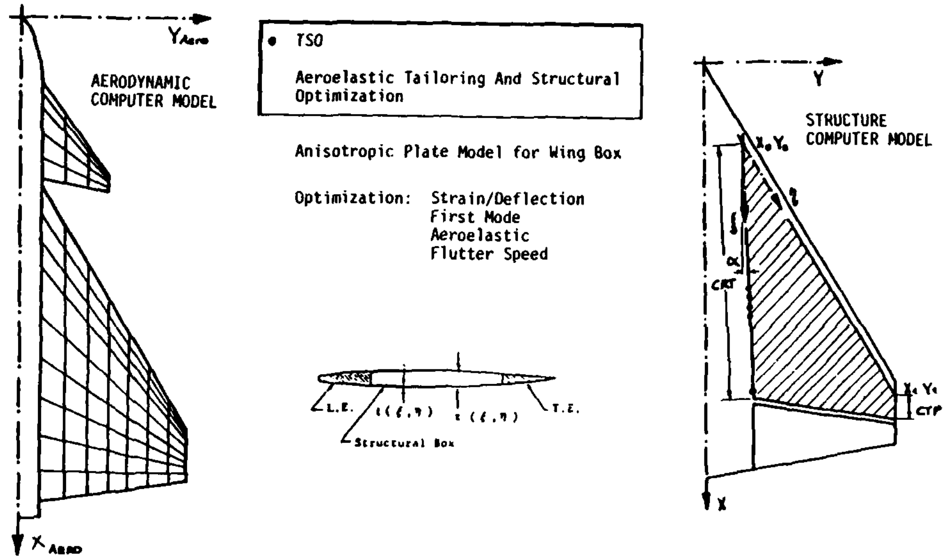


FIG. 3 COMPONENT ANALYSIS AND OPTIMIZATION WITH TSO

TSO structural input requires polynominal coefficients for the wing box depth (profil thickness) and laminat thickness distributions. A three dimensional graphic system has also subroutines to provide these polynominal coefficients.

A second advantage of TSO is the capability of aeroelastic tailoring. Mike Shirk et. al. have formulated a definition of aeroelastic tailoring in Ref. 5, which will be repeated here.

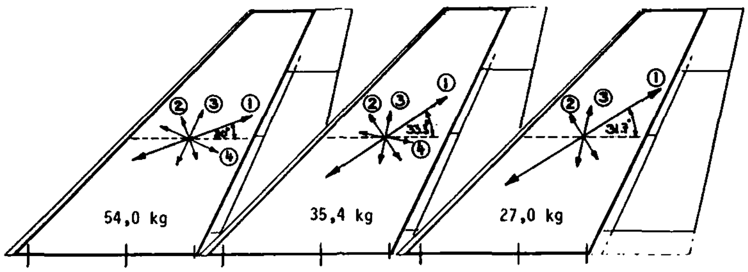
> "Aeroelastic tailoring is a design technique in which directional stiffness is used in aircraft structural design to control aeroelastic deformation, static or dynamic, in such a way that the aerodynamic and structural performance is affected in a beneficial way."

Two examples for a beneficial use of TSO and other optimization programs for aeroelastic design studies will be presented in the following chapters.

Fin Box Design Studies

The fin of a fighter aircraft has to fullfill static aeroelastic requirements for stability reasons and flutter requirements for an unfavourable mass distribution caused by a radar pod at the tip region. In Fig. 4 three different fin box laminates are shown.

The 54 kg version was already 'engineered' and now used as an initial design and starting point for a TSO optimization. After several runs TSO found a 35.4 kg box version which still meets design requirements. The third solution which is given in Fig. 4 shows a TSO optimization result which is already questionable due to local stress and fabrication reasons, but it shows in an impressive way the abilities of aeroelastic tailoring. The information about an optimal fibre orientation and the percentage of the main layers is valuable for further studies.



Fin Box Skin Weight	54 kg	35.4 kg	27 kg
Main Fibre Angle θ	20°	33.3°	31.7°
Layer ① 0°	19.7 kg 41%	16.4 kg 46.4%	16.8 kg 61.6%
Weight ② 90°	7.4 kg 15.4%	6.6 kg 18.6%	4.9 kg 18.1%
③ +45°	10.5 kg 21.8%	6.2 kg 17.5%	5.5 kg 20.3%
④ -45°	10.5 kg 21.8%	6.2 kg 17.5%	-
Fin Efficiency	0.77	0.81	0.83
Rudder Efficiency Ma 1.8 800 kts	0.48	0.52	0.52

FIG. 4 DIFFERENT FIN BOX LAMINATES

After these design studies with a plate type structure, other investigations were carried out to solve a rather severe flutter case caused by the fundamental fin bending mode and a low frequency torsion mode. This torsion mode was initiated by a local mass concentration (radar pot) at the fin tip region. No improvements in flutter behaviour could be achieved by adding balance masses. Only a flutter optimization with Grumman FASTOP Program (Flutter and Strength optimization procedure) on a finite element model gave a reasonable solution for this problem. A summary of all results of fin design studies are shown in Fig. 5.

AEROELASTIC

WEIGHT OPTIMIZATION WITH EFFICIENCY CONSTRAINT (Ma 1.8/800 kts)

DESIGN : 155 kg
AFTER OPT. : 131 kg
WEIGHTSAVING : 18 %

FLUTTER

INCREASE OF FLUTTER SPEED WITH MINIMUM WEIGHT

DESIGN : V_F = 944 kts
AFTER OPT. : V_F = 1021 kts

PERCENTAGE

FLUTTER SPEED : 8.2 %
WEIGHT INCREASE : 2.8 %

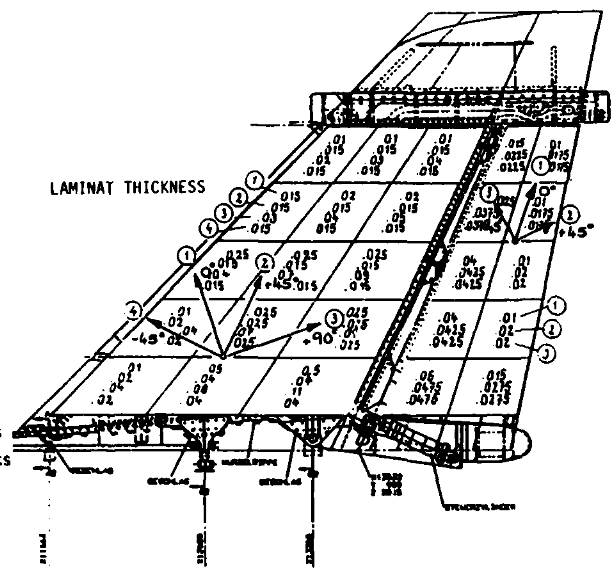
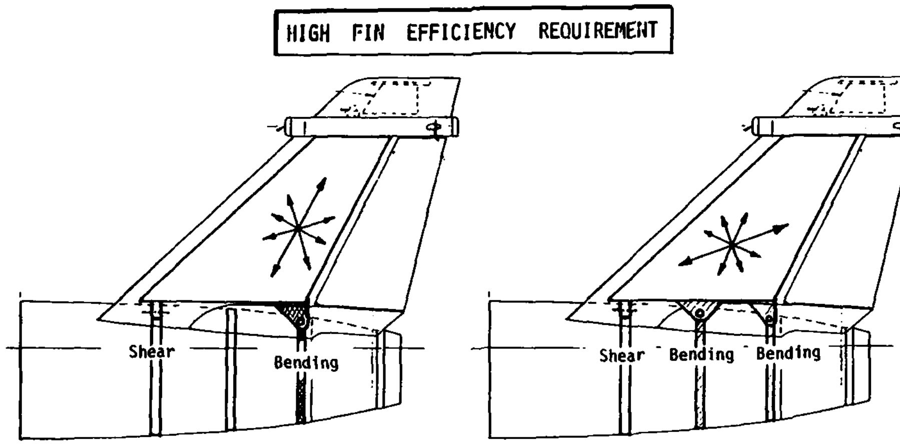


FIG. 5 SUMMARY OF FIN DESIGN STUDIES

One very important fact of these design studies should be pointed out. During component design studies different possibilities of attachment systems should be considered. In this fin study two different designs of a fin-fuselage connection were discussed. First a statically determinate support with a shear pick up at leading edge root and a bending pick up at the end of the fuselage in combination with a conservative fin box laminate.

Second an aeroelastic design with tailored laminate, one shear and two bending fittings, was suggested too.



STRESS DESIGN AEROELASTIC DESIGN

FIG. 6 TWO DIFFERENT FIN-FUSELAGE CONNECTION DESIGN

Both design concepts had similar component properties, but considering fuselage flexibilities, the aeroelastic design with the unusual fibre orientation has more advantages because of the second bending pick up.

Wing Box Design Studies

A similar application of TSO during a wing box design study will be presented in a brief way. A wing of a highly maneuverable fighter has to fullfill strong rollrequirements. Therefore the influence of wing box laminates had to be found out for different design philosophies. In Fig. 7 three different designs are compared due to skin weight, rollmoment efficiency, balanced and unbalanced laminate.

	Total Skin Weight lbs	Rollmoment Efficiency Ma 1.2/S.L.	Fibre Angle θ_i	Layer Weight Percentage
OPTIMIZATION STRESS ONLY BALANCED LAMINAT	395.5	0.05	76.3° θ_1 + 45° θ_1 - 45°	0.567 0.217 0.217
OPTIMIZATION STRESS ROLLEFFICIENCY BALANCED LAMINAT	518.1	0.29	71.3° θ_2 + 45° θ_1 - 45°	0.455 0.272 0.272
OPTIMIZATION STRESS ROLLEFFICIENCY UNBALANCED LAMINATE	464.6	0.28	79.3° θ_2 + 45° θ_1 - 45°	0.392 0.129 0.486

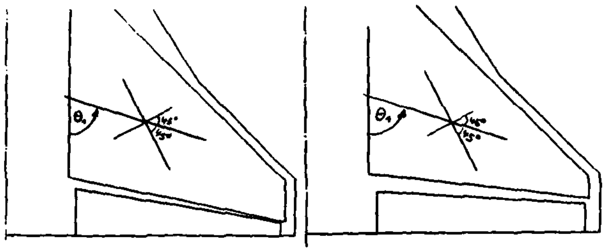
FIG. 7 COMPARISON OF WING DESIGN STUDIES

There is a remarkable relation in skin weight and stiffness requirement (roll efficiency). The stress designed wing-flap configuration had only low rollmoment efficiency at the most important design point at Ma 1.2/Sealevel. A satisfactory roll effectiveness could only be achieved by increasing skin weight about 30%, using conservative balance laminate. An unbalanced laminat would only require a weight increase of 17%.

Applied aeroelastic tailoring can be very helpful and provides important informations in the early design phase, because later changes are costly and time consuming.

Another application which should be mentioned, is a general wing flap geometry study. Roll capabilities depend very much on the wing flap geometry. Geometry variations are possible in a rather short time with TSO program because the structural input is simple.

A comparison of two different flap geometries is given in Fig. 8. This study shows that an increase of the flap tip chord will increase the elastic rollmoment by 12% with any weight penalty on the wing box.



	FLAP TIP CHORD 20 %	FLAP TIP CHORD 40 %
TOTAL SKIN WEIGHT	389.2 lbs	380.3 lbs
FIBRE ANGLE	θ_1 = 73.2° θ_1 + 45° θ_1 - 45°	θ_1 = 72.2° θ_1 + 45° θ_1 - 45°
PERCENTAGE WEIGHT	0.433 0.152 0.415	0.461 0.183 0.356
ROLLEFFICIENCY Ma 1.2/S.L.	0.2695	0.2710
RIGID ROLLMOMENT DERIVATIVE	1.22537	1.38527
ELASTIC LOSS	-0.89295	-1.01201
ELASTIC ROLLMOMENT DERIVATIVE	0.33242	0.37326 (+ 12.3 %)

FIG. 8 WING FLAP GEOMETRY STUDY

All results of these aeroelastic design studies with TSO must be checked with more detailed structural computer models. In most cases this model will be a finite element model, but there is also a large variety in model fineness. It depends very much on the background and experience of the structural analyst which model he will use at a certain development phase.

In our fin design study we compared three different structural models, see Fig. 9.

These models, the plate type structure of TSO and two finite element models with coars and fine mesh showed a very good agreement in static aeroelastic analysis. The elastic spanwise loading for fin and rudder are well comparable. The structural representation of an aerodynamic surface for a total aircraft analysis should describe the elastic behaviour with a minimum of degrees of freedom, otherwise it would be difficult to keep the aeroelastic analysis for a complete aircraft manageable.

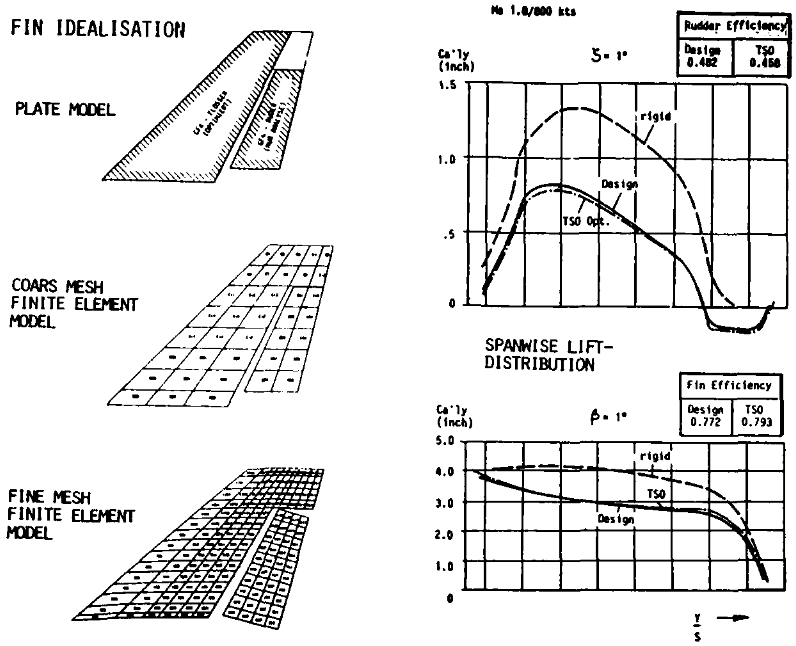


FIG. 9 COMPARISON OF DIFFERENT STRUCTURAL MODELS

AEROELASTIC ANALYSIS AND TESTING

Aeroelastic Analysis in preliminary Design

All mathematical models which are used in aeroelastic design studies include short comings and uncertainties. Therefore it is essential in static aeroelastic analysis to provide test datas for comparison and matching the computer models. According to the technical progress, there are different possibilities for comparison with aerodynamic and structural tests. A summary of work packages in aeroelastic analysis and testing during the development of an aircraft is given in Fig. 10.

	ANALYSIS	TESTING
PRELIMINARY DESIGN	Aeroelastic Tailoring Simplified total Aircraft Analysis Flexible Functions	Quasi Rigid Windtunnel Models Derivatives, Pressure Measurements Flexibility Corrections
DEVELOPMENT PHASE	Refined Aerodynamic and Structure Computer Models	Aeroelastic Windtunnel Models Composite Structures
FLIGHT CLEARANCE	Up-dated Computer Models Check Analysis	Static Deflection Measurement Ground Resonance Test Structural Coupling Test
FLIGHT TEST QUALIFICATION	Aerodynamic Data Set Improvements Closing the Design Loop	Lift and Drag Polars Aerodynamic Derivatives

FIG. 10 STATIC AEROELASTIC WORK PACKAGE

One of the activities in preliminary design is the first total aircraft analysis. After the analysis and optimization of aerodynamic surfaces it may cause difficulties to get a simplified fuselage structure for the total aircraft analysis. Structural analysis models of fighter aircraft fuselage are very complex with many degrees of freedom which takes a long time to create it. To overcome this problem a proposal for a preliminary fuselage model will be given. Using a mass distribution and a rough estimation of fuselage bending stiffness based on a similar aircraft, a free-free vibration analysis can be performed. If we assume a first bending mode frequency, the fuselage stiffness can be scaled to this frequency.

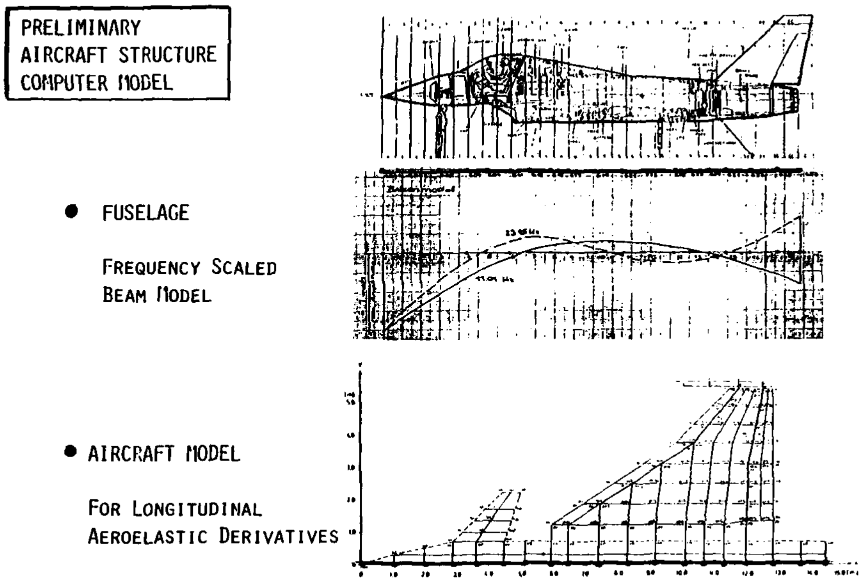


FIG. 11 SIMPLIFIED STRUCTURAL FUSELAGE MODEL

This very simple fuselage model has been statically coupled with a finite element model of wing, flap and foreplane for a symmetric total aircraft model. Using this structural model in combination with a two dimensional aerodynamic model the first flexibility function for longitudinal motions have been analyzed. The rigid and elastic pressure distribution for Ma 1.2 and unit angle of attack is plotted in Fig. 12. The ratio of elastic to rigid aerodynamic derivative is called effectiveness and this value depends on Mach Number and dynamic pressure.

The flexibility function of a trailing edge flap effectiveness is also shown in Fig. 12.

Aerodynamic derivatives for control surfaces of high maneouverable aircraft have an outstanding importance. Therefore great care must be taken in evaluation test results of 'quasi rigid' windtunnel models for matching the aerodynamic computer model. Thin wing profiles will cause already considerable flexibility effects in the high dynamic pressure region of windtunnel test.

Fig. 13 shows the finite element model of a windtunnel model with pick-up structure and the test rig for the measurement of influence coefficients. These deflection measurements were used to correct the structural computer model. The aeroelastic analysis gave important informations about the influence of windtunnel model flexibilities on measured aerodynamic datas.

Aeroelastic Analysis in Development Phase

One of the activities during development phase is the preparation of refined aerodynamic and structural computer models. It is not always possible to get a stiffness-matrix for the total aircraft as it is needed for the special purpose of aeroelastic analysis. In most cases stiffness and flexibility matrices for components are available, especially if different aircraft companies are working together.

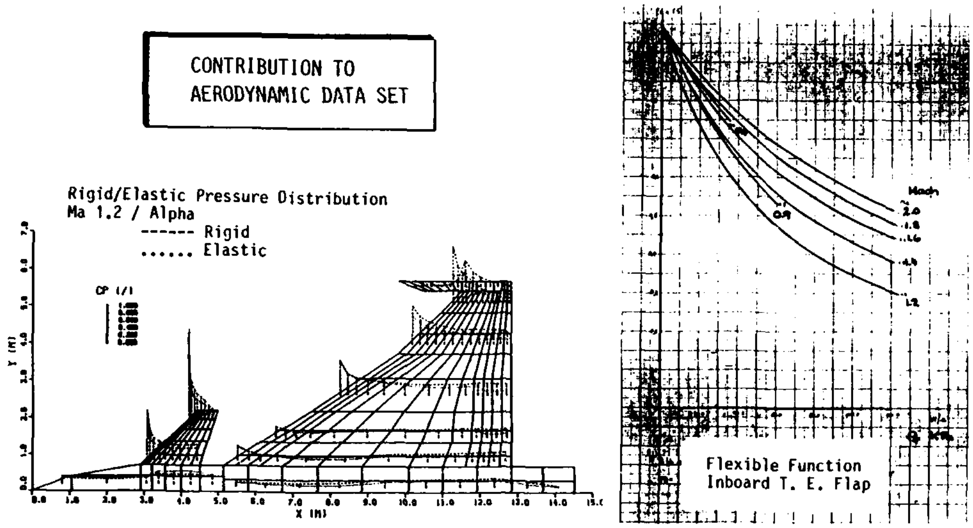


FIG. 12 PRELIMINARY LONGITUDINAL AEROELASTIC MODEL

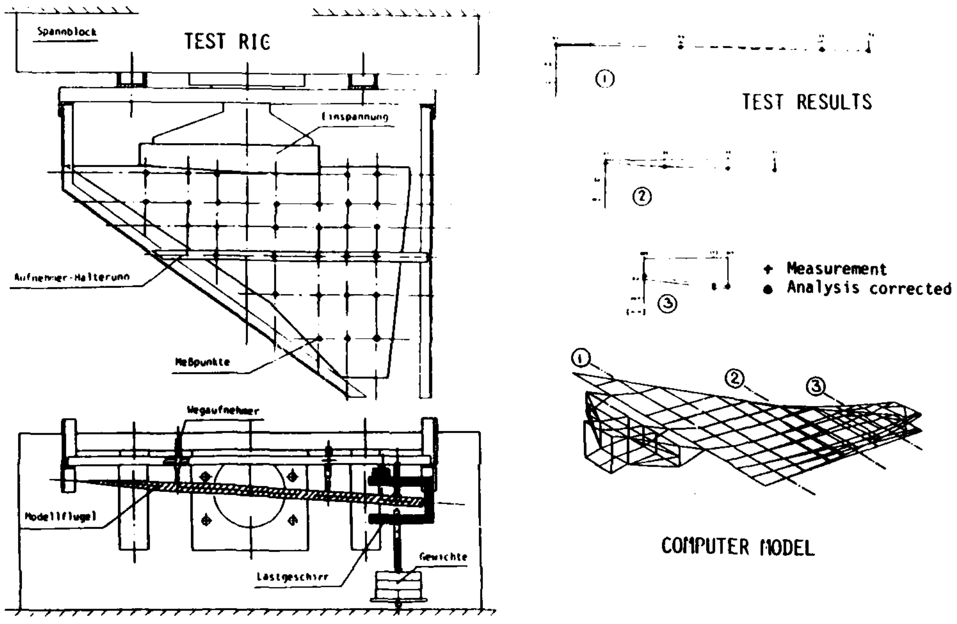


FIG. 13 WINDTUNNEL MODEL FLEXIBILITY

NASTRAN program has some useful capabilities to built up such a stiffness matrix from different components. The principle of this method is shown in Fig. 14

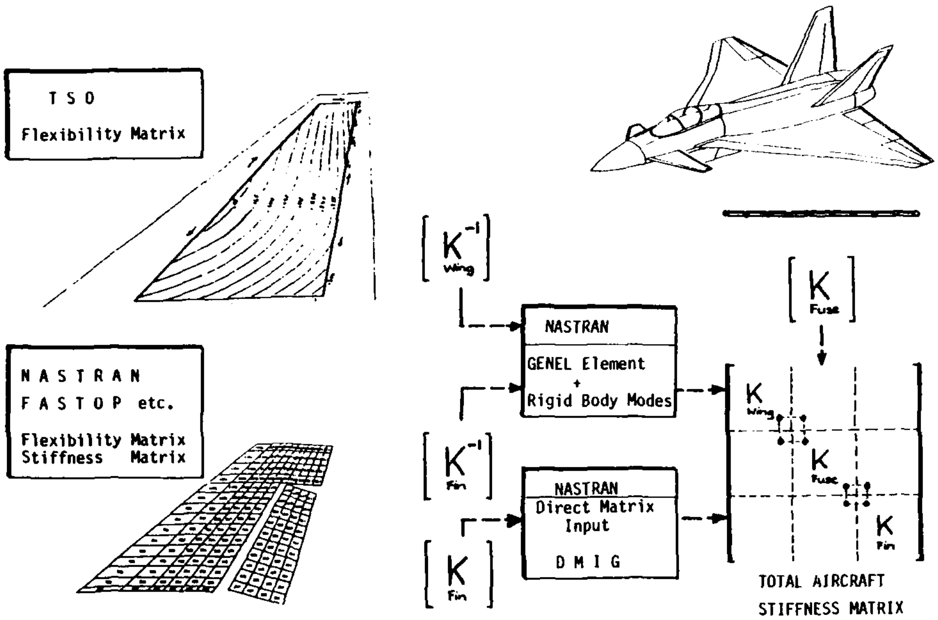


FIG. 14 STRUCTURAL AIRCRAFT MODEL

The reduction of degrees of freedom in the structural model must be consistent with the fineness of the aerodynamic model. Fig. 15 shows the structural model of a fighter aircraft with about 27000 degrees of freedom. The solution of this problem was carried out by a substructure technique and this procedure reduced the active degrees considerably.

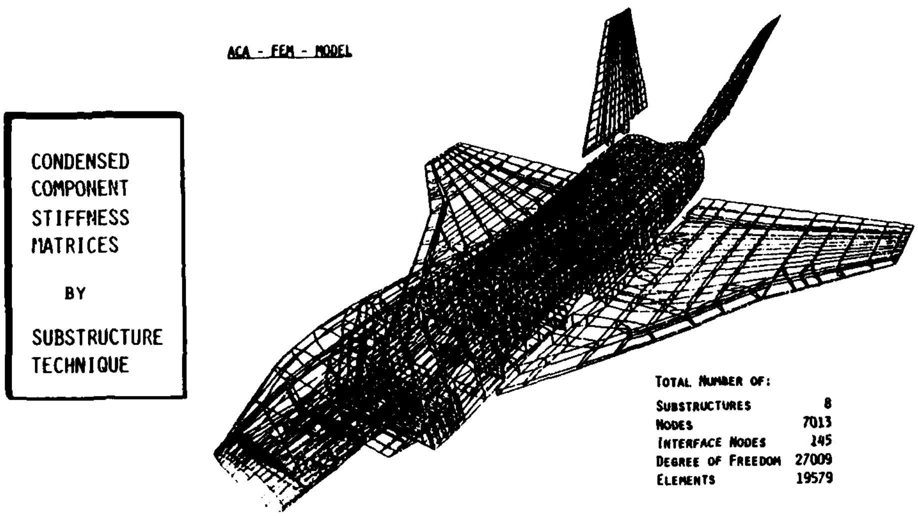


FIG. 15 SUBSTRUCTURES FOR TOTAL AIRCRAFT MODEL

Similar to the structural model, the aerodynamic model will change according to the actual development phase. A two dimensional model with about 200 panels will be sufficient for longitudinal motions in preliminary design. For lateral motions a three dimensional model is necessary because of the interference of wing and fin. This model can be built up with a reasonable number of about 200 panels. These two models and a much more sophisticated aerodynamic model is plotted in Fig. 16.

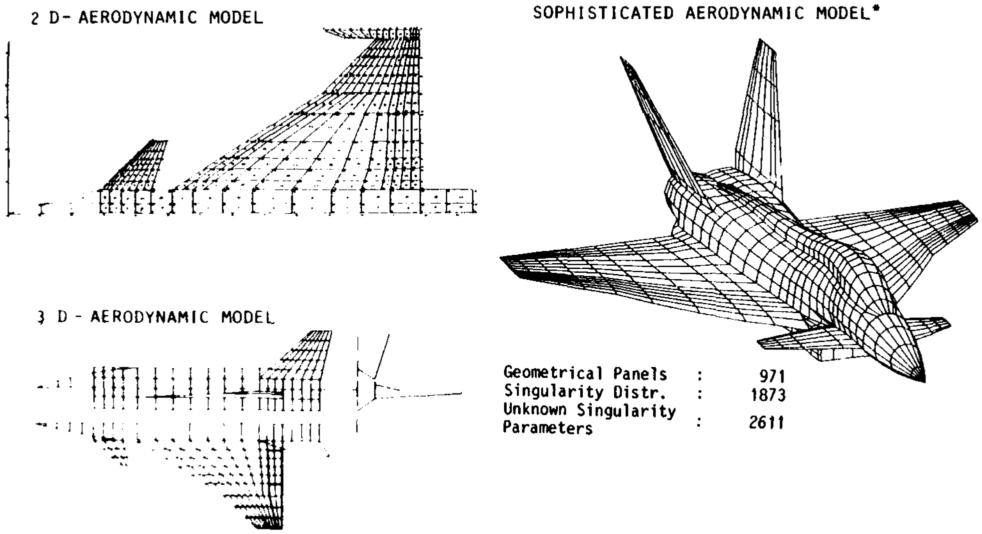


FIG. 16 AERODYNAMIC AIRCRAFT MODELS

[illegible] the sophisticated model is given in Ref. 7. This model is already [illegible] investigation and some kind of condensation must be done. The [illegible] parameter and no aerodynamic influence coefficient matrix [illegible] a direct solution of the aeroelastic equation system is not [illegible] be changed to an iterative solution. An advantage of the [illegible] possibility to match the aerodynamic loads on the structure [illegible] pressure measurements.

[illegible] system at a delta foreplane configuration with an angle of [illegible] analyzed in a water tunnel.

[illegible] between the measurement and the theoretical analysis shows consider- [illegible] New methods must be developed to use pressure distributions at [illegible] attack for interpolation to satisfy the aeroelastic equilibrium.

[illegible] loadings are very important for design load cases at high angle [illegible] possibility to correct the aerodynamic loads would be to factor each [illegible] aerodynamic model which is described in Ref. 6.

[illegible] analyst must decide which method he will use because it depends [illegible] available.

[illegible] idea is to built flexible windtunnel models, even aeroelastic [illegible] measure pressure distributions. At MBB a research program has been [illegible] scaled windtunnel models of a fin and the use of photogrammetry for [illegible] measurements will be tested. Two very good papers dealing also with these [illegible] in Ref. 6 and 8.

Test to Theory Comparison

Structural computer models for static aeroelastic calculations are similar to dynamic models and therefore we have already some possibilities for test to theory comparison before the first flight. The elastic behaviour of the total aircraft can be matched by normal mode shapes of the ground resonance test or frequencies of a structural coupling test.

The dynamic model of the Airbus, another highly qualified aircraft, is a combined finite element and stick model. Front fuselage, outer wing, outer vertical tail are idealized as beams. All the other parts of the aircraft are idealized as FE-structures.

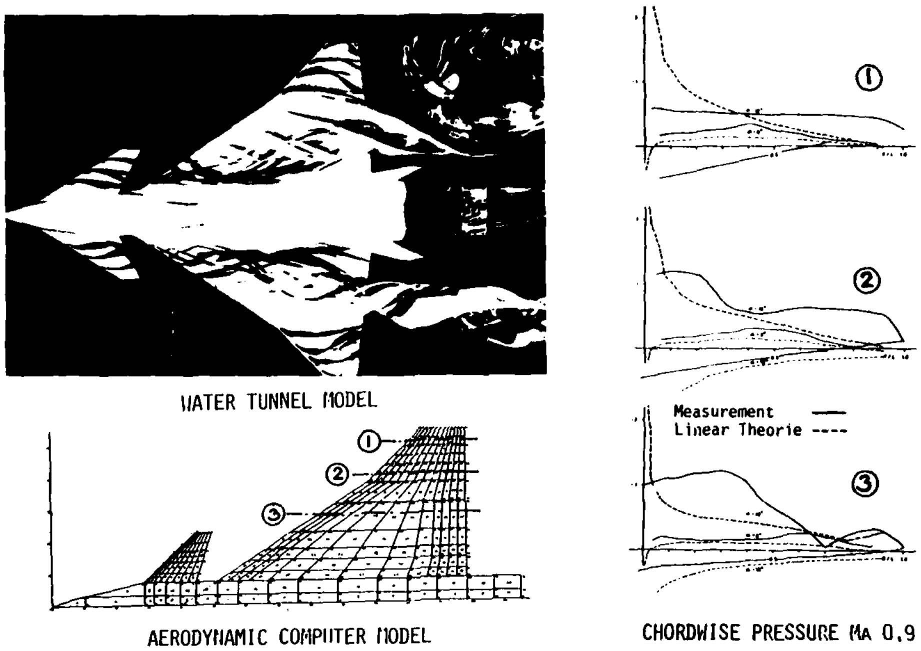
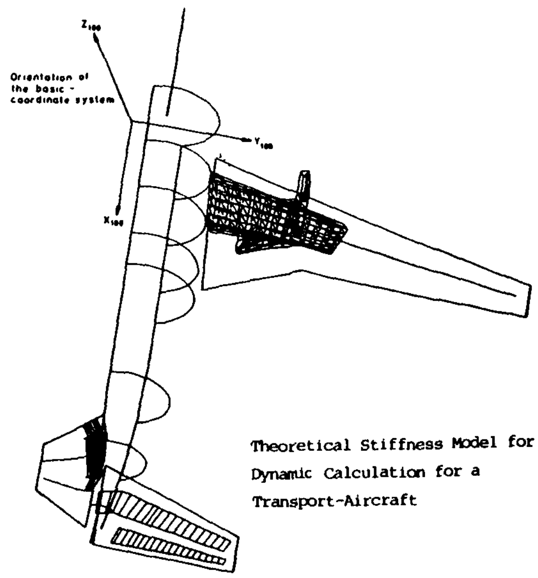


FIG. 17 WINDTUNNEL MODEL PRESSURE MEASUREMENT

To obtain a reasonable matching of the vibration calculation results with the GVT-results stiffness correction factors must be applied to the different parts of the stiffness model. They are shown in Fig. 18.



Stiffness Correction-Factors applied on the Theoretical Model

INNER WING	FEM stiffness	1,1
PYLON	in Z-direction	0,9
PYLON	in Y-direction	0,7
OUTER WING	in plane bending stiffness	∞
HORIZONTAL TAIL	FEM stiffness	1,2

FIG. 18 DYNAMIC MODEL OF AIRBUS

The correction factors that were applied to the pylon affect not only the pylon stiffness itself, but also its attachment stiffnesses between wing and pylon, and engine and pylon. That means the corrections applied to the pylon are necessary to improve the pylon and/or its attachment stiffnesses. The factor 0.9 applied to the stiffness of the pylon in z-direction means the torsion stiffness of the inner wing need not be changed but the bending stiffness must be increased by 10%. For the large correction necessary for the in-plane-bending stiffness there exists no plausible physical reason. The large correction factor necessary for the FE-structure of the horizontal tail can be decreased by a better idealization of the tail and its attachments.

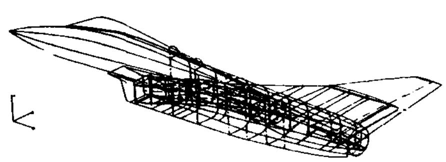
With these correction factors applied to the different parts of the theoretical model the vibration calculation results agree well with GVT-results for eigenfrequencies, mode shapes and generalized masses of the degrees of freedom up to the outer torsion mode of the wing and the first torsion mode of the horizontal and vertical tailplane. The theoretical prediction of total aircraft elasticity is difficult because it is not possible to calculate the local attachment stiffness of aircraft components good enough. Dynamic computer models contain the component and attachment or junction flexibilities in a separated form, and by matching these models to ground resonance tests these flexibilities can be determined.

Findly the flight test evaluation will provide a lot of datas for checking total aircraft aerodynamics and aeroelastic predictions, see Ref. 9.

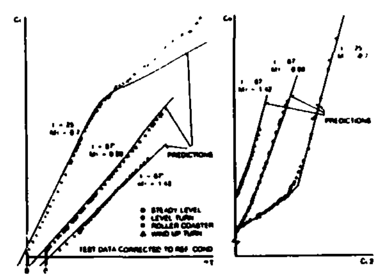
CONCLUDING REMARKS

The design philosophy of a high performance fighter aircraft has changed a lot during the past fifteen years. Structural and aerodynamic analysis and optimization programs have provided a design and analysis capacity which enables the aeroelastic analyst to begin with aeroelastic investigations already in the preliminary design phase.

These investigations must also include aeroelastic tailoring studies for lifting surfaces. Theoretical predictions and test results must be compared to match the mathematical models during the whole development and flight periode.



PRELIMINARY DESIGN



FLIGHTS TEST RESULTS •

- **Aeroelastic Design Studies as soon as possible**
- **Component and Total Aircraft Studies**
- **Comparison between Theoretical Prediction and Test Results**
- **Closing the Design Loop by Flight Test Evaluation**

FIG. 19 CONCLUSIONS

Full benefits of all new technologies can only be guaranteed by the right coordination between the different design disciplines, see Ref. 10.

REFERENCES

/1/ Aeroelastic Considerations in Preliminary Design of a Military Combat Aircraft;
M. Ormerod, D.G. Gibson; AGARD Conference Proceedings No. 354

/2/ Preliminary Design of Aircraft Using Structural Optimization Methods;
H. Gödel, G. Schneider, H. Hörnlein; AGARD-Conference Proceedings No. 354

/3/ Aeroelastic Considerations for Automatic Structural Design Procedures;
G. Schneider, H. Gödel; DGLR-Report 82-01

/4/ Aeroelastic Tailoring Studies in Fighter Aircraft Design;
W.E. Triplett; AIAA 79-0725

/5/ A Survey of Aeroelastic Tailoring, Theory, Practica, Promise;
M.U. Shirk, T.J. Hertz, T.A. Weisshaar; AIAA-84-0982

/6/ Static Aeroelastic Effects on the Aerodynamics of the SAAB 37 Viggen Aircraft,
A Comparison between Calculations, Windtunnel Tests and Flight Tests;
J. Kloos, L. Elmeland; ICAS-Paper 74-55

/7/ Calculation of Supersonic Flow Over Realistic Configrations by a Updated Law-Order Panel Method;
L. Fornasier; AIAA-83-0010

/8/ Wind Tunnel Test and Aerodynamic Analysis of three Aeroelastically Tailored Wings;
W.W. Braymon, W.A. Rogers, M.H. Shirk; ICAS-82-5.7.3

/9/ A Technique to Determine Lift and Drag Polars in Flight;
A. Knaus; AIAA-81-2420

/10/ Closing the Design Loop on HIMAT;
T.W. Putnam, M.R. Robinson; NA-84-1893

SECONDARY CONSIDERATIONS OF STATIC AEROELASTIC EFFECTS ON HIGH-PERFORMANCE AIRCRAFT

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SUMMARY

The flexibility inherent in high-performance aircraft has suggested this brief review of secondary effects in static aeroelasticity that heretofore have been regarded as negligible. Five topics are considered:

- redistribution of induced drag
- dihedral effect under load factor
- structural axis rotations in equations of motion
- aeroelastic divergence of an unrestrained vehicle
- corrections to measurements on "rigid" wind tunnel models

The few publications on these topics that have appeared over the years are surveyed on the assumption that secondary effects on earlier designs may become primary effects on future configurations.

REDISTRIBUTION OF INDUCED DRAG

The induced drag is obviously a primary concern for performance and any significant redistribution among components will affect total drag. A redistribution on a single component may also be of interest particularly as it may affect actuator requirements on a swing-wing configuration.

Only a few static aeroelastic wind-tunnel tests have been conducted over the years,[1-11] although they have been more frequent recently because of the possibilities for aeroelastic tailoring with composite materials (surveyed in Ref. 12) not only to stabilize a forward-swept-wing but also to improve drag characteristics. Only four tests are known to the author in which aeroelastic effects on the drag polar were investigated.

An elementary analysis of the total drag was given in Ref. 13. This study was motivated by the surprising performance of the North American F-107A airplane: the maximum flight speed exceeded its predicted value! The elementary analysis showed no significant change in the drag polar of the wing and only a small change in the total trim drag. The North American investigation then turned to further study of parasite and wave drag. A drag polar measured[2] on a low-speed aeroelastic (flutter) model of the Douglas XA3D-1 airplane is shown in Fig. 1 for an intermediate horizontal tail setting (i_t = +1 deg; settings of i_t = -2 deg, +4 deg, and tail-off were also measured). Since only the wing was flexible on the model, these data are consistent with the conclusion of Ref. 13 that the component drag polar is not affected by flexibility.

The Transonic Aircraft Technology (TACT) program had as its objective efficient aerodynamic performance both in cruise and during limit load maneuvering (7.33 load factor). The aeroelastic wind tunnel model of the TACT design[4] had the desired spanwise twist distribution in the level flight condition and the maximum washout twist in the limit load factor condition. Fig. 2 compares drag polars among rigid and flexible TACT models.

A U.S. Air Force Flight Dynamics Laboratory contract to General Dynamics Corp. had as its objectives to obtain wind-tunnel data on tailored wing designs and to demonstrate the attainable range of beneficial aeroelastic response by tailoring for both washin and washout.[11] The lift and drag characteristics at the design condition are shown in Fig. 3. The design for washout reduces tne drag due to lift primarily because it delays separation near the wing tip at the high design lift coefficient.

The highly maneuverable advanced technology (HiMAT) remotely piloted research vehicle, designed and built by Rockwell International for NASA Dryden Flight Research Center, was designed and built with an aeroelastically tailored canard and outboard wing. Its development program included a flexible model test to determine aeroelastic effects at high angles of attack.[5] The drag data obtained are similar to those shown in Figs. 2 and 3, but are the combined results of aeroelastic effects on two flexible components.

The results in Figs. 2 and 3 differ from those in Fig. 1 and the conclusions of Ref. 13 in that the component drag polar is affected by flexibility in the later tests. The problem warrants further investigation. The distribution of induced drag on lifting surfaces has been studied extensively at subsonic speeds.[14–22] Lan's quasi-vortex-lattice method[21,22] appears to be the most accurate method proposed for the calculation of subsonic induced drag. The Ames Wing-Body Program[18] used in Ref. 11 requires an empirical leading-edge-suction correction and is therefore limited in its utility.

DIHEDRAL EFFECT UNDER LOAD FACTOR

The dihedral effect is one of the many parameters that determines the lateral-directional dynamic stability of an airplane in the spiral and Dutch roll modes of motion. It also changes substantially because of symmetrical wing bending during maneuvers, e.g., in pullups or wind-up turns, that approach limit load factor. This aeroelastic effect has not been addressed in textbooks on aircraft stability and control.

The dihedral effect can be written in terms of its contributing sources as

$$C_{\ell_\beta} = (C_{\ell_\beta})_o + (\Delta C_{\ell_\beta})_t + (\Delta C_{\ell_\beta})_\Gamma + (\Delta C_{\ell_\beta})_{n_z} \qquad (1)$$

where $(C_{\ell_\beta})_o$ is the rigid dihedral effect arising from all sources other than geometric dihedral of the wing (e.g., wing-fuselage interference, wing sweep, wing tip shape, fin, stabilizer, etc.), $(\Delta C_{\ell_\beta})_t$ is the change in dihedral effect due to aeroelastic effects on the tail, $(\Delta C_{\ell_\beta})_\Gamma$ is the change in dihedral effect caused by the aeroelastic amplification of wing geometric dihedral, and $\Delta(C_{\ell_\beta})_{n_z}$ is the change caused by symmetrical wing bending and torsion. The term $(\Delta C_{\ell_\beta})_\Gamma$ has been considered by Seckel.[23]

The first analysis of $(\Delta C_{\ell_\beta})_{n_z}$ was given by Lovell.[24] It was based on lifting line theory in incompressible flow, a wing bending deflection curve of arbitary degree in the spanwise coordinate, and assumed that the wing tip deflection was proportional to the load factor n_z by an amount determined by the spanwise distributions of aerodynamic lift and bending stiffness. For the situation in which no detailed lift and stiffness distributions are known, Lovell proposed a deflection relationship based on a fully stressed design. Applications were made to a hypothetical fighter and a hypothetical bomber, and Lovell's results are reproduced in Table 1. The aeroelastic increments at limit load factor for the fighter and the bomber are seen to be 58% and 92%, respectively, of the rigid values of C_{ℓ_β}.

A later but independent solution was obtained by Rodden[25] after completing a correlation study[26] of the static aeroelastic tests of the Douglas XA3D-1 airplane results in Ref. 2. Assuming a parabolic deflection curve, Rodden showed a proportionality between $(\Delta C_{\ell_\beta})_{n_z}$ and the damping-in-roll coefficient C_{ℓ_p} which could be obtained from published charts for various planforms. Rodden later summarized[27] the above methods and wind tunnel results and proposed an influence coefficient formulation of the problem. The experimental and predicted results are reproduced in Fig. 4. The lateral-directional tests were run at two geometric angles of attack, α_g = +1 deg and -3 deg. The wing of the XA3D-1 had an incidence relative to the fuselage of +4 deg since it was designed for catapult launching from an aircraft carrier, so with an angle of attack of α_g = -3 deg, there was not much lift on the wing. In Fig. 4, we see a large variation of C_{ℓ_β} with dynamic pressure at α_g = +1 deg, but not much variation for α_g = -3 deg, and we see a reasonable agreement with the simplified prediction.

The influence coefficient method for predicting $(\Delta C_{\ell_\beta})_{n_z}$ has never been implemented by the author in a computer solution. The critical ingredient in its implementation is a spanwise differentiation. This can be accomplished by using the surface spline of Harder and Desmarais.[28] This spline is based on an analytical solution for an infinite uniform plate over multiple supports, so it can be differentiated analytically, as has already been done in the streamwise direction for the aeroelastic addition to NASTRAN.[29]

Another aspect of the deflected wing is the change in pitching moment that arises from the drag that modifies the balancing tail load. Both the parasite drag and the induced drag acting on the symmetrically deflected wing lead to a nonlinear trim problem that can be solved by iterative methods. A stability analysis of the coupling between wing bending and drag also shows the possibilities of aeroelastic divergence and flutter.[30–32]

STRUCTURAL AXIS ROTATIONS IN EQUATIONS ON MOTION

The equations of motion of a flexible vehicle relate the applied loads to the motion of the mean axes. If the flexibility is expressed in terms of structural influence coefficients (SICs) of the structure restrained in a statically determinate manner, a set of structural axes can be defined by the restraint configuration. If the structural influence coefficients are modified to include the inertial relief effects of the unrestrained vehicle, then the flexibility is obtained relative to the mean axes, and the equations of motion, expressed in terms of this free-body flexibility matrix, lead to correct solutions for maneuvering and dynamic response of the vehicle. This is the basis of the FLEXSTAB computer program.[33,34] If the flexibility matrix is not modified for inertial relief effects, any transient solution to the equations of motion is not invariant with the choice of restraint system for the SICs and is, therefore, not correct. However, an alternative formulation of the equations of motion, that determines the motion of the structural axes without modification of the SICs, has been obtained[35] by accounting for the rotations of the structural axes relative to the mean axes. The correctness of this alternative formulation was demonstrated numerically in maneuvering studies of a simplified forward-swept-wing airplane. The differences between solutions that considered the rotations between the structural and mean axes and those that did not were not dramatic for the example airplane, but it was noted that the differences would be expected to be configuration dependent.

The simplified forward-swept-wing (FSW) airplane example of Ref. 35, which was analyzed by strip theory and is shown in Fig. 5, has been reconsidered in Ref. 36 using the subsonic doublet-lattice lifting surface theory and assuming a wing incidence angle of +1.0 deg relative to the fuselage. The derivatives for the example FSW airplane are reproduced in Table 2 for sea level flight at Mach 0.9. The restrained values of the derivatives are for the SICs restrained at the intersection of the fuselage and wing elastic axis; the unrestrained values of the derivatives correspond to the FLEXSTAB formulation.

AEROELASTIC DIVERGENCE OF AN UNRESTRAINED VEHICLE

A consideration of the relative angle between the longitudinal structural and mean axes also provided the basis for correcting an incorrect solution to the divergence problem of an unrestrained vehicle. In Ref. 37, Rodden identified a singularity in the FLEXSTAB stability derivatives as quasi-steady divergence of the vehicle in free-flight. In the Errata,[38] it was concluded, by following a derivation of Letsinger,[39] that the singularity does not correspond to a physical divergence, but is only a mathematical property of the mean longitudinal axis. In the FLEXSTAB formulation the lift curve slope C_{z_α} and angle of attack α give the lift coefficient $C_z = C_{z_\alpha} \alpha$ where α is the angle of attack between the free-stream and the longitudinal mean axis. The coefficient C_{z_α} becomes singular when the structure deforms in such a manner that the mean axis remains aligned with the free-stream, i.e., when α remains zero but C_z remains finite.

The foregoing difficulty might be regarded as a consequence of performing a static stability analysis on a problem that requires a dynamic stability analysis. The British flutter method was applied to a problem of unrestrained aeroelastic divergence in Ref. 40. The simple example consisted of a two-dimensional two-degree-of-freedom airfoil mounted on a "fuselage" free only in plunge. The analysis illustrated the interesting characteristic of the British flutter method that at certain speeds all roots correspond to mechanical degrees of freedom but at other speeds some of the roots correspond to aerodynamic lags, as had been shown earlier.[41] In addition, the analysis obtained the surprising result that the divergence root, which was real for the restrained system, appeared as complex in the unrestrained case with a mass ratio (wing mass divided by total mass) r = 0.50. The authors chose to call this root "dynamic divergence" rather than flutter because the instability finds its origin in the tendency to static divergence. One might have assumed that with a lower mass ratio (i.e, a heavier fuselage) the dynamic divergence would become a static divergence again, but this did not occur for mass ratios as low as r = 0.01; the divergence root remains oscillatory but with decreasing frequency as the mass ratio is reduced. Two sets of results of dampings and frequencies for velocities up to 300 ft/s are shown in Fig. 6 and for r = 0.25, and in Fig. 7 for r = 0.05.

CORRECTIONS TO MEASUREMENTS ON "RIGID" WIND TUNNEL MODELS

The small-scale solid steel wind tunnel model is usually regarded as rigid. However, larger scale models of advanced configurations may no longer permit the assumption of rigidity, and aeroelastic corrections to the measurements would then become necessary.

A wind tunnel model of the high-aspect-ratio wing of the Northrop X-21, the laminar flow control research airplane, was designed in 1960 with interchangeable wing root airfoil sections to study wing-fuselage interference. This design required a small spar to support the outboard wing panel. Unfortunately, the small spar introduced a great deal of flexibility into the wing and the test became an aeroelastic rather than aerodynamic test, and all of the data required aeroelastic corrections. An aeroelastic computer program[42] was modified to obtain the inverse solution for the rigid load distribution in terms of the measured lifting pressures via the equation

$$\{C_{z_r}^{(e)}\} = [B_m]^{-1}\{C_{z_f}^{(e)}\} \tag{2}$$

where $\{C_{z_f}^{(e)}\}$ is the set of force coefficients measured on the flexible model, $\{C_{z_r}^{(e)}\}$ is the set of desired force coefficients on the rigid configuration, and $[B_m]^{-1}$ is the inverse of the model aeroelastic load amplification matrix given by

$$[B_m]^{-1} = [I] - (\bar{q}S/\bar{c})[C_{hs}][a] \tag{3}$$

where $[C_{hs}]$ and $[a]$ are the static aerodynamic influence coefficients (AICs) and structural influence coefficients (SICs), respectively, of the model, $\bar{q}$ is the test value of dynamic pressure, and S and $\bar{c}$ are the reference area and chord, respectively, of the model. [see also Eqs. (10) and (11b) of Ref. 35]. The subsonic AICs were obtained from the lifting surface theory of Runyan and Woolston.[43,44] The SICs of the wing were obtained at grid points convenient for the laboratory measurements and interpolated to the grid points of the aerodynamic theory[45]. The corrections were straight-forward, but the extensive requirements for data adjustment were regarded as excessive and new "rigid" models were constructed and returned to the wind tunnel to obtain data that were directly usable.

A 1971 wind-tunnel test of the aerodynamic load distribution on a NASA supercritical-wing research airplane configuration has been reported in Ref. 46. The wing deflections were measured but no aeroelastic corrections were introduced. Changes in the location of the transonic shock wave with changes in the dynamic pressure were believed to be associated with aeroelastic effects rather than Reynolds number effects. Estimates of wing stiffness of the full-scale airplane also suggested that some aeroelastic scaling existed so that wind-tunnel aerodynamic characteristics obtained at the higher dynamic pressures would be expected to approximate full-scale characteristics.

The problem of correcting aerodynamic data measured on flexible models may be routine if the aerodynamic behavior is linear (we assume the model flexibility will always be linear) or the corrections are small. With nonlinearities or large corrections, the linear method described above might be applied in an iterative manner, although it may introduce significant errors that warrant investigation, particularly when there is movement of the boundary layer separation or transonic shock waves. Perhaps further development of aeroelastic modelling technology is the only alternative to measure pressure distributions reliably.

CONCLUDING REMARKS

This paper has not only surveyed some lesser-known aspects of static aeroelasticity that are readily calculated but it has also suggested some topics that warrant further research. One of these is extending the methods of calculating the distribution of induced drag to account for wing-body interference. Another is improving techniques for minimizing the total drag in multiple flight conditions by aeroelastic optimization of camber and twist. Experimental work on aeroelastic models will be a necessary adjunct to these investigations.

The problem of correcting wind-tunnel measurements for the flexibility inherent in "rigid" models appears to be a formidable one at high angles of attack or transonic speeds. An iterative piecewise linear solution has been mentioned as an initial approach; it may suggest refinements to that approach or possibly another and better approach. The development of an aeroelastic pressure model technology should also be considered and might provide the only alternative to correction procedures that could prove to be unreliable.

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REFERENCES

1. Gaugh, W. J., and Slap, J. K. "Determination of Elastic Wing Aerodynamic Parameters," Journal of the Aeronautical Sciences, March 1952, pp. 173-182.

2. Ogawa, H. "Report of Wind-Tunnel Tests of a 1/12 Scale Aeroelastic Model of the Douglas (El Segundo) XA3D-1 Airplane," GALCIT Rept. 601, California Institute of Technology, Pasadena, CA, 30 April 1953.

3. Roskam, J., Holgate, T., and Shimizu, G. "Elastic Wind-Tunnel Models for Predicting Longitudinal Stability Derivatives of Elastic Airplanes," Journal of Aircraft, Nov.-Dec. 1968, pp. 543-550.

4. Waddoups, M. E., Smith, C. B., and Mc Mickle, R. W. "Composite Wing for Transonic Improvement; Vol. I - Composite Wing Aeroelastic Response Study," AFFDL-TR-71-24, Air Force Flight Dynamics Laboratory, Dayton, OH, Dec. 1972.

5. Sokolsky, O. M. "Analysis of HiMAT RPRV .22 Scale Flexible Model; Wind Tunnel Tests TWT-332 and TWT-335," Rept. No. TFD-79-09, Rockwell International, Los Angeles, CA, 8 Jan. 1979.

6. Wilkinson, K., and Rauch, F. "Predicted and Measured Divergence Speeds of an Advanced Composite Forward Swept Wing Model," AFWAL-TR-80-3059, Air Force Wright Aeronautical Laboratories, Dayton, OH, July 1980.

7. Ellis, J. W., Dobbs S. K., and Miller, G. D. "Structural Design and Wind-Tunnel Testing of a Forward Swept Fighter Wing," AFWAL-TR-80-3073, Air Force Wright Aeronautical Laboratories, Dayton, OH, July 1980.

8. Ricketts, R. H., and Doggett, R. V., Jr. "Wind-Tunnel Experiments on Divergence of Forward-Swept Wings," NASA Technical Paper 1685, Aug. 1980.

9. Blair, M., and Weisshaar, T. A. "Swept Composite Wing Aeroelastic Divergence Experiments," Journal of Aircraft, Nov. 1982, pp. 1019-1024.

10. Sherrer, V. C., Hertz, T. J., and Shirk, M. H. "Wind Tunnel Demonstration of Aeroelastic Tailoring Applied to Forward Swept Wings," Journal of Aircraft, Nov. 1981, pp. 976-983.

11. Braymen, W. W., Rogers, W. A., and Shirk, M. H. "Wind Tunnel Test and Aerodynamic Analysis of Three Aeroelastically Tailored Wings," Paper No. ICAS-82.5.7.3, 13th Congress of the International Council of the Aeronautical Sciences, Seattle, WA, 22-27 Aug. 1982.

12. Shirk, M. H., Hertz, T. J., and Weisshaar, T. A. "A Survey of Aeroelastic Tailoring-Theory, Practice, Promise", AIAA Paper No. 84-0982-CP, AIAA/ASME/ASCE/AHS 25th Structures, Structural Dynamics and Materials Conference, Palm Springs, CA, 14-16 May 1984.

13. Rodden, W. P., Surber, T. E., and Vetter, H. C. — "The Effect of Flexibility on the Drag Polar of an Aircraft," Journal of the Aeronautical Sciences, June 1957, pp. 456-457.

14. Rubbert, P. E. — "Theoretical Characteristics of Arbitrary Wings by a Non-Planar Vortex Lattice Method," Rept. D6-9244, The Boeing Co., Seattle, WA, 1964.

15. Garner, H. C., Hewitt, B. L., and Labrujere, T. E. — "Comparison of Three Methods for the Evaluation of Subsonic Lifting-Surface Theory," Rept. NLR TN-G.65, National Aerospace Laboratory, The Netherlands, June 1968.

16 Garner, H. C. — "Some Remarks on Vortex Drag and Its Spanwise Distribution in Incompressible Flow," Journal of the Royal Aeronautical Society, July 1968, pp. 623-625.

17. Wagner, S. — "On the Singularity Method of Subsonic Lifting-Surface Theory," Journal of Aircraft, Nov.-Dec. 1969, pp. 549-558.

18. Carmichael, R. L., Castellano, C. R., and Chen, C. F. — "The Use of Finite Element Methods for Predicting the Aerodynamics of Wing-Body Combinations," Paper No. 2, NASA SP-22, Symposium on Analytic Methods in Aircraft Aerodynamics, Ames Research Center, 28-30 Oct. 1969.

19. Kalman, T. P., Giesing, J. P., and Rodden, W. P. — "Spanwise Distribution of Induced Drag in Subsonic Flow by the Vortex Lattice Method," Journal of Aircraft, Nov.-Dec. 1970, pp. 574-576.

20. Kalman, T. P., Giesing, J. P., and Rodden, W. P. — "Reply by Authors to G. J. Hancock," Journal of Aircraft, Aug. 1971, pp. 681-682.

21. Lan, C. E. — "A Quasi-Vortex-Lattice Method in Thin Wing Theory," Journal of Aircraft, Sept. 1974, pp. 518-527.

22. Lan, C. E. — "The unsteady quasi-vortex-lattice method with applications to animal propulsion," Journal of Fluid Mechanics, Part 4, 1979, pp. 747-765.

23. Seckel, E. — Stability and Control of Airplanes and Helicopters, Academic Press, New York, 1964.

24. Lovell, P. M., Jr. — "The Effect of Wing Bending Deflection on the Rolling Moment due to Sideslip," NACA TN 1541, Feb. 1948.

25. Rodden, W. P. — "A Simplified Expression for the Dihedral Effect of a Flexible Wing," Journal of the Aeronautical Sciences, Aug. 1955, p. 579.

26 Rodden, W. P. — "Analysis and Summary of Low-Speed Aeroelastic Model Tests of the XA3D-1 Airplane," Douglas Aircraft Company, Inc., Rept. ES 17246, Feb. 1953.

27. Rodden, W. P. — "Dihedral Effect of a Flexible Wing," Journal of Aircraft, Sept.-Oct. 1965, pp. 368 -373.

28. Harder, R. L., and Desmarais, R. N. — "Interpolation Using Surface Splines," Journal of Aircraft, Feb. 1972, pp. 189-191.

29. Rodden, W. P., Harder, R. L., and Bellinger, E. D. — "Aeroelastic Addition to NASTRAN," NASA CR 3094, March 1979.

30. Petre, A. — Teoria Aeroelasticitiatii-Statica, Editura Academiei Republicii Socialiste Romania, Bucharest, 1966.

31. Goetz, R. C. — "Divergence of Some All-Movable Control Surfaces Including Drag Loadings," NASA TN D-4793, Oct 1968.

32. Petre, A., and Ashley, H. — "Drag Effects on Flutter," Journal of Aircraft, Oct. 1976, pp. 755-763.

33. Anon. — "An Analysis of Methods for Predicting the Stability Characteristics of an Elastic Airplane - Summary Report," NASA CR-73277, Nov. 1968.

34. Dusto, A. R., Brune, G. W., Dornfeld, G. M. Mercer, J. E., Pilet, S. C., Rubbert, P. E., Schwantz, R. C., Smutny, P., Tinoco, E. N., and Weber, J. A. — "A Method for Predicting the Stability Derivatives of an Elastic Airplane; Vol. I - FLEXSTAB Theoretical Description", NASA CR-114712, Oct. 1974; or AFFDL TR-74-91, Vol. 1, Nov. 1974.

35. Rodden, W. P., and Love, J. R. — "Equations of Motion of a Quasisteady Flight Vehicle Utilizing Restrained Static Aeroelastic Characteristics," Journal of Aircraft, Sept. 1985, pp. 802-809.

36. Rodden, W. P., Wilson, C. T., Herting, D. N., Bellinger, E. D., and MacNeal, R. H. — "Static Aeroelastic Addition to MSC/NASTRAN," Paper No. 15, MSC/NASTRAN Users' Conference Proceedings, Pasadena, CA, 22-23 March 1984.

37. Rodden, W. P. — "Aeroelastic Divergence of Unrestrained Vehicles," Journal of Aircraft, Dec. 1081, pp. 1072-1073.

38. Rodden, W. P. — "Errata - 'Aeroelastic Divergence of Unrestrained Vehicles'," Journal of Aircraft, Jan. 1984, pp. 94-96

39. Letsinger, G. R. — "Stability and Control Derivatives Including Aeroelastic Effects," unpublished Document No. D6A 11589-1TN, The Boeing Aerospace Co., Feb. 1969.

40. Rodden, W. P., and Bellinger, E. D. — "Unrestrained Aeroelastic Divergence in a Dynamic Stability Analysis," Journal of Aircraft, Sept. 1982, pp. 796-797.

41. Rodden, W. P., and Bellinger, E. D. — "Aerodynamic Lag Functions, Divergence, and the British Flutter Method," Journal of Aircraft, July 1982, pp. 596-598.

42. Rodden, W. P., Farkas, E. F., and Slack, F. C. — "Quasi-Steady Aero-Thermo-Elastic Analysis: Analytical Development and Procedure for the IBM 7090 Computer," Rept. NOR-61-58, Northrop Corp., 14 April 1961.

43. Runyan, H. L., and Woolston, D. S. — "Method for Calculating the Aerodynamic Loading on an Oscillating Finite Wing in Subsonic and Sonic Flow," NACA Rept. 1322, 1957.

44. Rodden, W. P., Hodson, C. H., and Revell, J. D. — "Subsonic and Sonic Aerodynamic Influence coefficients from Unsteady Lifting Surface Theory," Rept. NA-57-1104, North American Aviation, Inc., 25 Oct. 1957.

45. Rodden, W. P. — "Further Remarks on Matrix Interpolation of Flexibility Influence Coefficients," Vol. 26, No.11, Nov. 1959, pp. 760-761.

46. Harris, C. D. — "Wind-Tunnel Measurements of Aerodynamic Load Distribution on a NASA Supercritical-Wing Research Airplane Configuration," NASA TM X-2469, Feb. 1972.

Table 1. Increment in Rolling Moment Coefficent Due to Sideslip for Hypothetical Fighter-type and Bomber-type Airplanes

	Fighter	Bomber
Aspect ratio	6.0	10.0
Taper ratio	0.500	0.500
Dihedral angle, deg	5.00	4.00
Limit load factor	8.00	2.67
C_{l_β} for rigid wing	-0.0650	-0.0613
Increment C_{l_β} for load factor of 1	-0.0047	-0.0212
Increment C_{l_β} for one-half limit load factor	-0.0188	-0.0283
Increment C_{l_β} for limit load factor	-0.0376	-0.0566

Table 2. Derivatives for Example FSW Airplane

Derivative	Rigid Value	Restrained Value	Unrestrained Value
C_{L_o}	0.8422	0.10638	0.1337
C_{m_o}	0.06624	0.08523	0.10630
C_{L_α}	5.071	6.413	8.109
C_{m_α}	4.736	5.951	7.201
C_{L_δ}	0.2461	0.3177	0.4512
C_{m_δ}	0.9407	1.0663	1.1101
C_{L_q}	-3.140	-4.786	-7.480
C_{m_q}	-6.050	-7.649	-9.592
$C_{L_{\ddot{z}}}$	0.0	-0.002850	-
$C_{m_{\ddot{z}}}$	0.0	-0.005123	-
$C_{L_{\ddot{\theta}}}$	0.0	-0.008810	-
C_m	0.0	-0.02103	-
α_{p_o}	0.0	-0.0001383	-
α_{p_α}	1.0	1.0659	1.0
α_{p_δ}	0.0	0.07378	-
α_{p_q}	0.0	0.1915	-
$\alpha_{p_{\ddot{z}}}$	0.0	-0.003321	-
$\alpha_{p_{\ddot{\theta}}}$	0.0	-0.01680	-

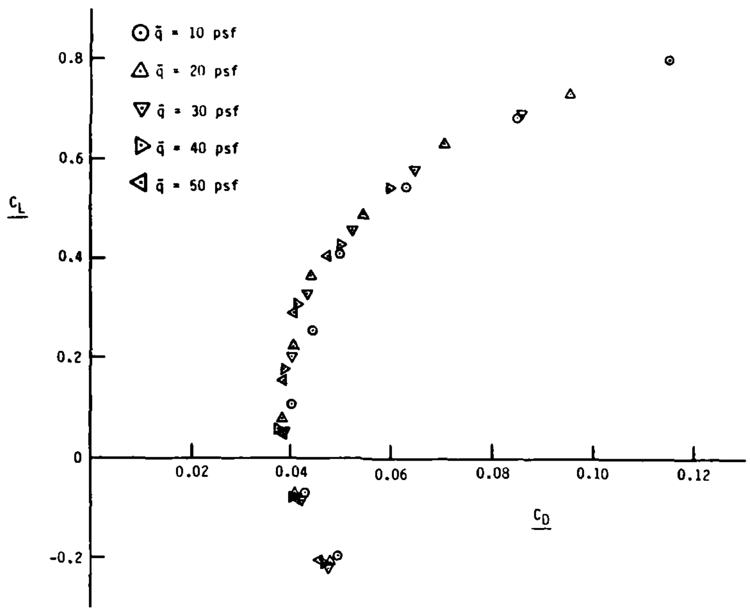


Fig. 1. Drag Polar of Douglas XA3D-1 Airplane with Flexible Wing.

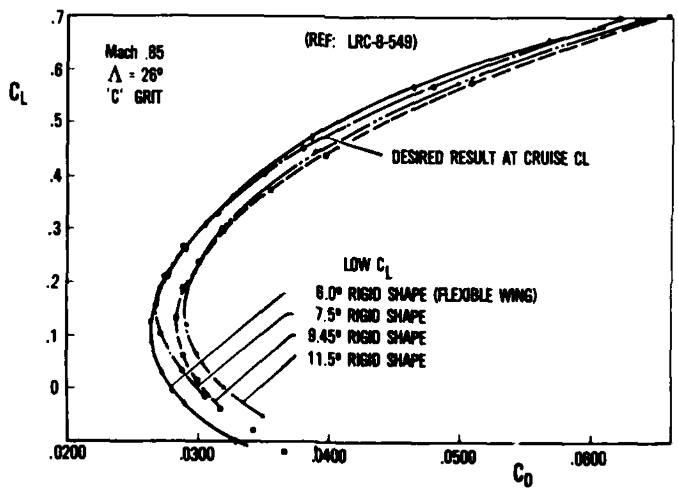
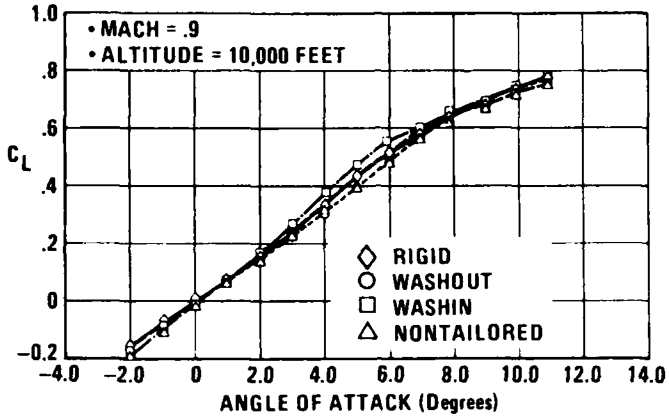
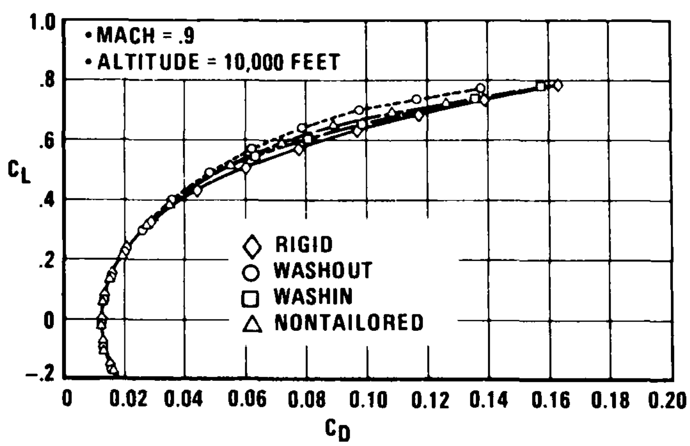


Fig. 2. Drag Polars of TACT Composite Aeroelastic and Rigid Models.



a. Lift Characteristics



b. Drag Characteristics

Fig. 3. Lift and Drag Characteristics of Aeroelastically Tailored Research Wings at Design Condition.

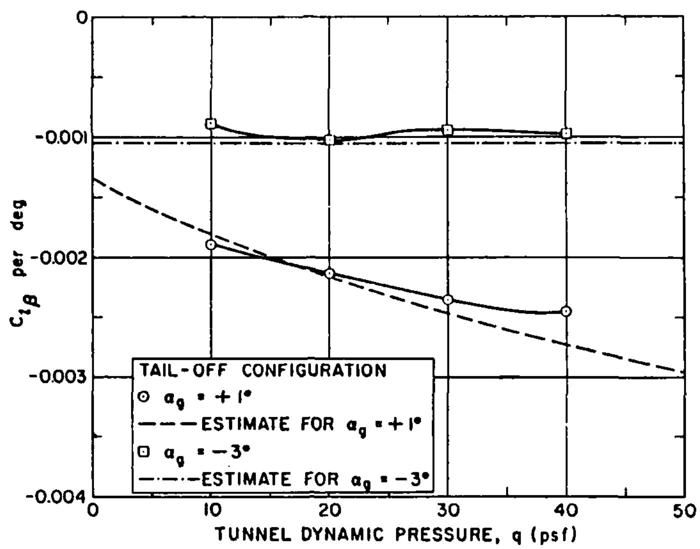


Fig. 4. Correlation of Predicted and Measured Values of Dihedral Effect on Douglas XA3D-1 Airplane with Flexible Wing.

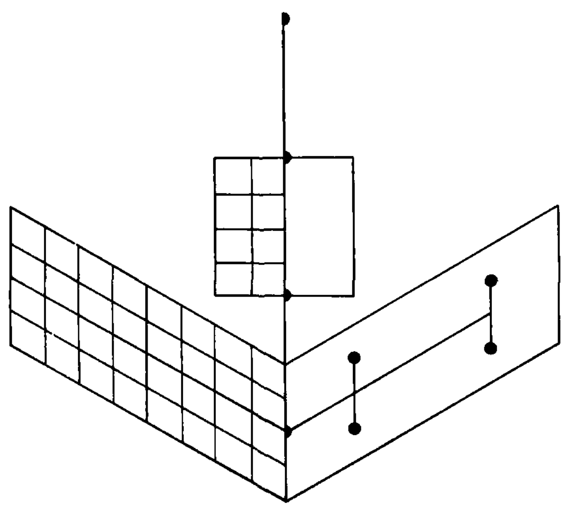


Fig. 5. Idealization of Forward-Swept-Wing Airplane Configuration for MSC/NASTRAN Aeroelastic Analysis.

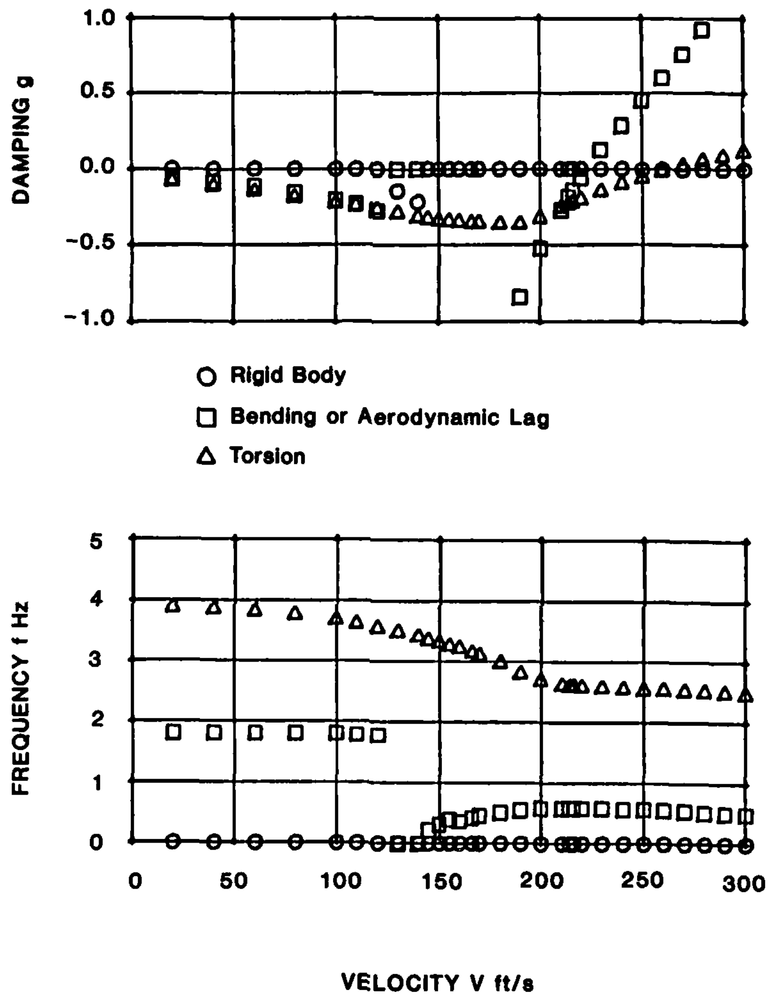


Fig. 6. Dampings and Frequencies of a 2 DOF Airfoil with Forward Centroid Plus a Fuselage Plunge Degree of Freedom with Mass Ratio r=0.25.

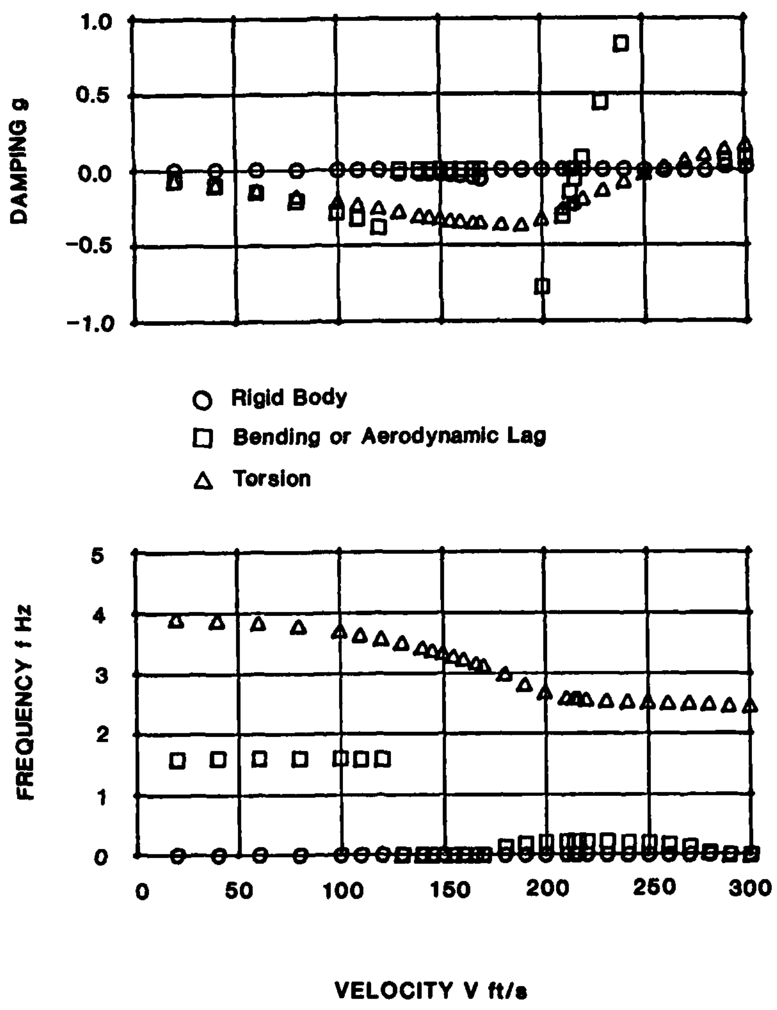


Fig. 7. Dampings and Frequencies of a 2 DOF Airfoil with Forward Centroid Plus a Fuselage Plunge Degree of Freedom with Mass Ratio r=0.05.

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The papers show that static aeroelasticity is increasingly being considered as a primary design parameter, affecting as it does structural optimisation, vehicle aerodynamic stability control effectiveness and overall performance.

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